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FOREWORD

This document contains references of papers published by the Langley Research Center in various areas of hypersonic aerodynamics for the period 1950-1986. The research work was performed either in-house by the Center staff or by other personnel supported entirely or in part by grants or contracts. Abstracts have been included with the references when available. The references are listed chronologically and are grouped under the following general headings:

1. AERODYNAMIC MEASUREMENTS-SIMPLE SHAPES
2. AERODYNAMIC MEASUREMENTS-CONFIGURATIONS
3. AERO-HEATING
4. CONFIGURATION STUDIES
5. PROPULSION INTEGRATION-EXPERIMENT
6. PROPULSION INTEGRATION-STUDY
7. ANALYSIS METHODS
8. TEST TECHNIQUES
9. AIRFRAME ACTIVE COOLING SYSTEMS

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SUBJECT CATEGORY 1 - AERODYNAMIC RESULTS - SIMPLE SHAPES

- 1. Becker, John V.: Results of Recent Hypersonic and Unsteady Flow Research at the Langley Aeronautical Laboratory. Journal of Applied Physics, vol. 21, no. 7, July 1950, pp. 619-628.**

The extremely high speeds and altitudes now contemplated in supersonic aircraft design involve many new compressible flow problems which, only a few years ago, were either unrecognized or of purely academic interest. This situation has naturally resulted in a greatly increased concentration of effort in basic compressible flow research at establishments such as the NACA. The purpose of this paper is to review briefly some recent results obtained at Langley in projects of this character dealing with hypersonic and unsteady flows. Similar exploratory work in small-scale facilities is in progress at the Ames and Lewis Laboratories of the NACA.

- 2. McLellan, C. H.: Exploratory Wind-Tunnel Investigation of Wings and Bodies at $M = 6.9$. Journal of Aeronautical Sciences, vol. 18, no. 10, October 1951.**

Results of an exploratory investigation of wings and bodies at a Mach Number of 6.9 in the Langley 11-inch hypersonic tunnel are presented. Force data were obtained on a square and a triangular plan-form wing having 5 per cent thick diamond sections, on a circular cone-cylinder body, and on a semicircular cone-cylinder body. Pressure distribution data were also obtained on the square plan-form wing and the circular body. The results have been compared with calculations made by various theoretical methods. The more exact theoretical treatments gave overall force values in good agreement with experimental results. Measured pressure distributions departed appreciably from calculated results in some cases because of the influence of boundary layer.

- 3. Cooper, Ralph D.; and Robinson, Raymond A.: An Investigation of the Aerodynamic Characteristics of a Series of Cone-Cylinder Configurations at a Mach Number of 6.86. NACA RM L51J09, December 1951.**

The results of pressure-distribution and force tests of two series of cone-cylinder configurations at a Mach number of 6.86 in the Langley 11-inch hypersonic tunnel are presented and compared with theoretical calculations. The first series, which consisted of three configurations, having 20° conical noses with apex angles varying from 10° to 180° , was tested in the axially symmetric attitude. In general, satisfactory agreement was obtained between experimental results and theoretical calculations.

- 4. Ulmann, Edward F.; and Bertram, Mitchel H.: Aerodynamic Characteristics of Low-Aspect-Ratio Wings at High Supersonic Mach Numbers. NACA RM53123, November 1953.**

This paper presents some recently obtained data on the aerodynamic characteristics of low-aspect-ratio wings at supersonic Mach numbers of 4.04 and 6.9 and discusses some new methods of predicting the lift and drag of such wings. Data on lifting wings in the Mach number range above 2.5 are not plentiful and most of the available data may be found in references 1 to 8.

5. **Penland, Jim A.: Aerodynamic Characteristics of a Circular Cylinder at Mach Number 6.86 and Angles of Attack up to 90°. NACA RM L54A14, March 1954.**

Pressure-distribution and force tests of a circular cylinder have been made in the Langley 11-inch hypersonic tunnel at a Mach number of 6.86, a Reynolds number of 129,000, and angles of attack up to 90°. The results are compared with the hypersonic approximation of Grimminger, Williams, and Young and a simple modification of the Newtonian flow theory. An evaluation of the crossflow theory is made through comparison of present results with available crossflow Mach number drag coefficients.

6. **Ridyard, Herbert W.: The Aerodynamic Characteristics of Two Series of Lifting Bodies at Mach Number of 6.86. NACA RM L54C15, May 1954.**

The results of force tests of two series of lifting bodies in the Langley 11-inch hypersonic tunnel at a Mach number of 6.86 and Reynolds numbers from 1.9×10^6 to 2.6×10^6 based on body length are presented and compared with theory. One series consisted of 10° cone cylinders with various afterbody lengths. The other series consisted of drooped-nose, flat-bottomed bodies with variations in fineness ratio, nose shape, and plan-form aspect ratio.

7. **Bertram, Mitchel H.: Viscous and Leading-Edge Thickness Effects on the Pressures on the Surface of a Flat Plate in Hypersonic Flow. Journal of Aeronautical Sciences, vol. 21, no. 6, June 1954, pp 430-431.**

8. **McLellan, Charles H.: A Method for Increasing the Effectiveness of Stabilizing Surfaces at High Supersonic Mach Numbers. NACA RM L54F21, August 1954.**

Large increases in lift-curve slope at low angles of attack at high supersonic Mach numbers can be obtained by the use of wedge-shape airfoil sections. The use of such sections on the tail surfaces operating at low angles of attack on airplanes or missiles traveling at these speeds can greatly decrease the stabilizing-surface area required. Moderate increases in section effectiveness (lift-curve slope) over that for more conventional tail sections can be obtained with little or no increase in total drag because of the decreased surface area required. Larger increases can be obtained; however, they will be accompanied by increases in stabilizing surface drag.

9. **Bertram, Mitchel H.; and McCauley, William D.: Investigation of the Aerodynamic Characteristics at High Supersonic Mach Numbers of a Family of Delta Wings Having a Double-Wedge Sections with the Maximum Thickness at 0.18 Chord. NACA RM L54G28, October 1954.**

The aerodynamic characteristics of a family of blunt double-wedge-section delta wings have been investigated at the Langley 11-inch hypersonic tunnel. These wings had a maximum thickness of 8 percent located at the 18-percent-chord point. For the wings tested at a Mach number of 6.9, the semiapex angle of attack from 0 to 28 and Reynolds numbers from 0.8 to 3.6×10^6 was based on root chord. Pertinent results from tests at Mach numbers as low as 1.6 have been utilized. The shock-expansion and Newtonian impact theories have been used to analyze the effects of changes made in the various parameters investigated.

10. **Fetterman, David E. and Ridyard, Herbert W.: The Effect of a Change in Airfoil Section on the Hinge-Moment Characteristics of a Half-Delta Tip Control with a 60° Sweep Angle at a Mach Number of 6.9. NACA RM L54H16a, October 1954.**

A theoretical and experimental investigation has been made of the effect of a change in airfoil section on the hinge-moment characteristics of a half-delta tip control with a 60° sweep angle. Shock expansion and linear theories were used in comparison with the test results of two tip controls of different airfoil section. The tests were made at a Mach number of 6.9 and a Reynolds number of 0.64×10^6 , based on tip control mean aerodynamic chord. The controls were investigated at angles of attack of 0° and 8° over a control-deflection range of -14° to 14°, and at zero control deflection over an angle-of-attack range of -12° to 12°.

11. **McCauley, William D.; and Feller, William V.: An Investigation of the Characteristics of the NACA RM-10 (With and Without Fins) in the Langley 11-Inch Hypersonic Tunnel at a Mach Number of 6.9. NACA RM L54103, November 1954.**

Pressure distributions on the body and force balance results for the NACA RM-10 with and without fins are presented for angles of attack from 0° to about 20° at $M = 6.9$. The experimental results are compared with the following theories: method of characteristics, Van Dyke's second-order theory, conical-shock two-dimensional expansion theory, Newtonian impact theory, Grimminger, Williams, and Young's correlation prediction, and linear theory. Interference between fins and body is discussed. The effect of Mach number on the components of zero-lift drag and of the pitching moment, center of pressure, and lift-curve slope at zero lift are presented.

12. **Penland, Jim A.: Aerodynamic Characteristics of a Circular Cylinder at Mach Number 6.86 and Angles of Attack up to 90°. NACA RM L54A14, 1954.**

Pressure-distribution and force tests of a circular cylinder have been made in the Langley 11-inch hypersonic tunnel at a Mach number of 6.88, at Reynolds number of 129,000, and angles of attack up to 90°. The results are compared with the hypersonic approximation of Grimminger, Williams, and Young and a simple modification of the Newtonian flow theory. An evaluation of the crossflow theory is made through comparison of present results with available crossflow Mach number drag coefficients.

13. **Bertram, Mitchel H.; and McCauley, William D.: An Investigation of the Aerodynamic Characteristics of Thin Delta Wings With a Symmetrical Double-Wedge Section at a Mach Number of 6.9. NACA RM L55B14, April 1955.**

The aerodynamic characteristics of thin delta wings with symmetrical double-wedge sections have been investigated in the Langley 11-inch hypersonic tunnel at a Mach number of 6.9. A family of 5-percent-thick lifting wings with semiapex angles varying from 30° to 8° and one wing which has a 2-1/2 percent thickness and a semiapex angle of 8° were tested over a range of angle of attack from 0 to 35. The 5-percent-thick and a series of 2-1/2 percent thick wings were tested at zero lift to evaluate the effects of Reynolds number and thickness ratio. The range of Reynolds numbers for these tests was from 0.7 to 5.6×10^6 . Results have been compared with shock-expansion and Newtonian impact theories.

14. **Feller, William V.: Investigation of Equilibrium Temperatures and Average Laminar Heat-Transfer Coefficients for the Front Half of Swept Circulars Cylinder at a Mach Number of 6.9. NACA RM L55F08a, August 1955.**

Average heat-transfer coefficients and equilibrium temperatures for the front half of an isothermal cylinder with a laminar boundary layer were determined from wind-tunnel tests at Mach number 6.9, free-stream Reynolds numbers based on diameter of 1.3×10^5 and 1.8×10^5 , and sweep angles from 0° to 75° . The reductions of average heat-transfer coefficient and equilibrium temperature with sweep angle were compared with simple theories. The laminar correlating parameter, when the air conductivity and viscosity are evaluated at the temperature just behind the bow shock, was found to vary only slightly with sweep angle.

15. **Beckwith, Ivan E.: Theoretical Investigation of Laminar Heat Transfer on Yawed Infinite Cylinders in Supersonic Flow and a Comparison With Experimental Data. NACA RM L55F09, August 1955.**

An integral method is presented for the calculation of the compressible laminar boundary layer on yawed infinite cylinders of arbitrary shape and with arbitrary wall temperature distribution. General expressions are derived for the heat transfer and recovery factor in the vicinity of the stagnation line. The method is applied to the calculation of local heat-transfer rates on a yawed circular cylinder at a stream Mach number of 6.9. The results of this calculation are in good agreement with some experimental data for small angle of yaw but at a yaw angle of 60° the theoretically predicted heat transfer is about 40 percent less than the measured values.

16. **McLellan, Charles H.; and Dunning, Robert W.: Factors Affecting the Maximum Lift- Drag Ratio at High Supersonic Speeds. NACA RM L55L20a, February 1956.**

A study of the factors affecting the maximum lift-drag ratio $(L/D)_{MAX}$ has been conducted in an effort to determine how to obtain high aerodynamic $(L/D)_{MAX}$ values at high supersonic Mach numbers. Wings, bodies, and wing-body combinations are discussed, and some of the effect of leading-edge heating on wing geometry and $(L/D)_{MAX}$ are included. It appears hopeful that high $(L/D)_{MAX}$ values may be achieved at the high supersonic Mach numbers by utilization of as high a Reynolds number laminar flow as possible, low-aspect-ratio wings favorable interference effects, and the use of more radical configurations.

17. **Ulmann, Edward F.; and Ridyard, Herbert W.: Flow-Field Effects on Static Stability and Control at High Supersonic Mach Numbers. NACA RM L55L19a, March 1956.**

Recent wind-tunnel investigations of aircraft-type configurations at Mach numbers 4.06 and 6.86 have provided data which show that flow-field interference is of primary importance in stability and control calculations at high supersonic Mach numbers and that the location of stabilizing and control surfaces to give highest effectiveness can be determined by theoretical studies of these flow fields. A method has been derived which predicts the trend of downwash around a circular body as the angle of attack is increased. A method has also been derived which gives good predictions of the tail contributions to lateral stability through a considerable angle-of-attack range.

18. **Penland, Jim A.; and Fetterman, David E., Jr.: The Effects at a Mach Number of 6.86 of Drag Brakes on the Lift, Drag, and Pitching Moment of an Ogive Cylinder. NACA RM L55K23, March 1956.**

Three-component force tests of an ogival cylinder having a fineness ratio of 9.5 equipped with two diametrically opposite fuselage panel-type drag brakes have been made in the Langley 11-inch hypersonic tunnel at a Mach number of 6.86 and a Reynolds number of 1.5×10^6 based on body length. The drag brakes were 1-1/2 body diameters long and 0.60 body diameters wide and were tested at brake deflection angles from 0° to 30° in horizontal and vertical positions at body angles of attack from -5° to 25° . The comparison of experimental results with the predictions of Newtonian impact theory shows that the trends of the longitudinal characteristics with angle of attack may be predicted with reasonable accuracy. The drag brakes in the vertical position produce higher total drag and higher negative pitching moments at angles of attack than do the identical brakes in the horizontal position, even though the top drag brake becomes ineffective at high angles of attack.

19. **Ridyard, Herbert W.; and Fetterman, David E.: Aerodynamic Characteristics of a 6-Percent-Thick Symmetrical Circular-Arc Airfoil Having a 30-Percent Chord Trailing-Edge Flap at a Mach Number of 6.9. NACA RM L56B24, June 1956.**

An investigation, including pressure distributions and schlieren flow photographs, has been made of the flow characteristics over a 6-percent-thick symmetrical circular-arc airfoil section with a 30-percent-chord trailing edge flap at a Mach number of 6.90 and a Reynolds number of 1.65×10^6 . The model was tested over an angle-of-attack range of 0° to 16° and a flap deflection range of -16° to 16° . The experimental pressure distributions are in good agreement with the results of shock-expansion theory except in the regions of flow separation (on the low-pressure side of the flap and in the vicinity of the hinge line) which result from shock-boundary-layer interaction.

20. **Bertram, Mitchel H.: Tip Bluntness Effects on Cone Pressures at $M = 6.85$. Journal of the Aeronautical Sciences, vol. 23, no. 9, September 1956.**

The purpose of the present investigation is to examine the aerodynamic effect of a simple type of nose blunting on a basic body. The body chosen was a 10° half-angle cone with a base diameter of 2.00 in., and the blunting consisted of a plane cut normal to the cone axis. This is similar to the blunting previously investigated by this writer for the two-dimensional case. The model was fitted with five rows of pressure orifices. There was one orifice at the first station, two orifices at the second and fourth stations, and four orifices at the third and fifth stations. Successive blunting of this cone supplied a large number of data points. The tests were conducted in the nominal Mach 7 Invar nozzle of the Langley 11-inch hypersonic tunnel. Because of variations in pressure and the times at which data were taken during a run, the calibration Mach Number varied between 6.84 and 6.87, the value of 6.85 being a good average for all the data. The data were obtained at supply pressure in the range 22 to 34 atmospheres and at supply temperatures between 1065°R. and 1170°R. , and the corresponding Reynolds Numbers per inch were in the range 2.4×10^6 to 3.8×10^6 .

21. **Ridyard, Herbert W.; Dunning, Robert W.; Johnston, E. W.: Aerodynamic Characteristics from Wind-Tunnel Studies of the X-15 Confirmation. Research Airplane Committee Report on Conference on the Progress of the X-15 Project, Langley Aeronautical Lab., October 1956.**

In order to investigate the aerodynamic characteristics of the X-15 research airplane, an exploratory wind-tunnel test program was initiated in January of 1956. Since that time, X-15 models have been tested in eight different facilities through a Mach number range from less than 0.1 to about 6.9. Several variations of the original configuration have been tested. The aerodynamic characteristics of two of the configurations are presented in this paper.

22. **Feller, William V.: Method of Calculation of Heat Transfer Coefficients for the X-15 Airplane. Research Airplane-Committee Report on Conference on the Progress of the X-15 Project. October 25-26, 1956, pp. 113-126.**

The wind-tunnel results presented in this paper have been compared with available theories to give some indication of how well the theories can be expected to predict the heat transfer to the full-scale airplane. In general, for isolated parts of the airplane which can be approximated by simple shapes, the heat transfer can be satisfactorily predicted by available theories. For regions where there is interference between the flows on adjacent parts - for example, the wing-fuselage juncture, cockpit canopy, and side fairings - more detailed studies are required on the specific configuration, and for this purpose a complete scale model is being prepared by North American Aviation for wind-tunnel tests in the Langley Unitary Plan wind tunnel.

23. **Penland, Jim A.: Aerodynamic Characteristics of a Circular Cylinder at Mach Number 6.86 and Angles of Attack Up to 90°. NACA TN 3861, January 1957 (Supersedes NACA RM L54A14, 1954).**

Pressure-distribution and force tests of a circular cylinder have been made in the Langley 11-inch hypersonic tunnel at a Mach number of 6.88, a Reynolds number of 129,000, and angles of attack up to 90°. The results are compared with the hypersonic approximation of Grimminger, Williams, and Young and a simple modification of the Newtonian flow theory. An evaluation of the crossflow theory is made through comparison of present results with available crossflow Mach number drag coefficient.

24. **Love, Eugene S.: Base Pressure at Supersonic Speeds on Two-Dimensional Airfoils and on Bodies of Revolution With and Without Fins Having Turbulent Boundary Layers. NACA TN 3819, January 1957.**

An analysis is made of available experimental data to show the effect of most of the variables that are more predominant in determining base pressure at supersonic speeds. The analysis is restricted to turbulent boundary layers and covers two-dimensional bases and the bases of bodies of revolution with and without stabilizing fins. An analogy to the peak pressure rise associated with the separation of the boundary layer is presented as simple semiempirical methods for the estimation of base pressure.

25. **McLellan, Charles H.; Bertram, Mitchel H.; and Moore, James A.: An Investigation of Four Wings of Square Plan Form at a Mach Number of 6.9 in the Langley 11-inch Hypersonic Tunnel. NACA Report 1310, 1957. (Supersedes NACA RM L51D17, 1951).**

The results of tests of four wings at a Mach number of about 6.86 and a Reynolds number of 0.98×10^6 are presented. The wings had square plan forms with 5-percent-thick diamond, half-diamond, wedge, and half-circular-arc sections. The boundary layer was found to have a large effect on the pressure distributions. Reasonable agreement was indicated between the aerodynamic coefficient from experimental data.

26. **Bertram, Mitchel H.: Exploratory Investigation of Boundary-Layer Transition on a Hollow Cylinder at a Mach Number of 6.9. NACA Rep. 1313, 1957. (Supersedes NACA TN 3546.**

27. **Henderson, Arthur, Jr.; and Bertram, Mitchel H.: Factors Affecting Loads at Hypersonic Speeds. NACA RM L57D25a, June 1957.**

This paper gives a brief summary of current loads information at hypersonic speeds. Several methods which the designer can employ in estimating the loads on various aircraft components are discussed. The paper deals with the characteristics of both slender and blunt configurations and touches upon the effect of boundary-layer and aerodynamic interference.

28. **Bond, A. C.; Feller, W. V.; and Bland, W. M.: Aerodynamic Heat Transfer to Wing Surfaces and Wing Leading Edges, NACA RM L57D22C, June 1957.**

29. **Bertram, Mitchel H.; and Henderson, Arthur, Jr.: Effects of Boundary-Layer Displacement and Leading-Edge Bluntness on Pressure Distribution, Skin Friction, and Heat Transfer of Bodies at Hypersonic Speeds. NACA TN-4301, July 1958.**

Results are presented of an investigation to determine the effect of boundary-layer displacement and leading-edge bluntness on surfaces in hypersonic flow. The presence of the boundary layer and the blunt leading edge induce pressure gradients which in turn affect the skin friction and heat transfer to the surface. Methods for predicting these phenomena on two-dimensional surfaces are given and a brief review of recent three-dimensional results is presented.

30. **Penland, Jim A.; and Armstrong, William O.: Preliminary Aerodynamic Data Pertinent to Manned Satellite Reentry Configurations. (Also in NACA Conference on Aerodynamics, Moffett Field, California, March 1958). NACA RM L58E13a, July 1958.**

Some recent experimental data at $M = 6.9$, together with calculations made by the modified Newtonian and shock-expansion theories, are presented for a variety of aerodynamic shapes considered for use as manned reentry vehicles. These vehicles were grouped in three basic categories: nonlifting bodies, lifting bodies, and airplane-like vehicles. The results indicate that from aerodynamic consideration, all of these configuration types are suitable for consideration as manned reentry vehicles.

31. **Lauten, William T., Jr.; Levey, Gilbert M.; and Armstrong, William O.: Investigation of an All-Movable Control Surface at a Mach Number of 6.86 for Possible Flutter. NACA RM L58B27, May 1958.**

Tests were made on a dynamically and elastically scaled model of a sweptback, all-movable horizontal tail proposed for the North American X-15 airplane and on several configurations having lower stiffnesses. The spectrum of natural vibration frequencies indicates that the weakest configuration was one-fourth as stiff as the stiffest configuration. No flutter was obtained.

32. **Bertram, Mitchel H.: Boundary-Layer Displacement Effects in Air At Mach Numbers of 6.8 and 9.6. NASA TR R-22, 1959. (Supersedes NACA TN 4133, Feb. 1958).**

Measurements are presented for boundary-layer-induced pressure gradients on a flat plate in aircraft at a Mach number of 9.6 and for the drag of thin wings at a Mach number of 6.8 and zero angle of attack. The investigation was conducted in the Langley 11-inch hypersonic tunnel, and the pressure measurements at a Mach number of 9.6 were made in the presence of substantial heat transfer from the boundary layer to the wall.

33. **Lord, Douglas R.; Smith, Fred M.; and Penland, Jim A.: Aerodynamic Characteristics of Various Missile Configurations at Angles of Attack from 0° to 90° for Mach Numbers from 2 to 7. Presented at the 40th Meeting of the Bumblebee Aerodynamics Panel, Silver Springs, Maryland, May 12-13, 1959 (X66-91047).**

Several investigations were made at the NASA Langley Research Center to determine the effects of systematic changes in geometry on the performance and stability characteristics of various hypersonic missile configurations. This paper presents some of the experimental and theoretical results of three of these programs.

34. **Henderson, Arthur, Jr.; and Johnston, Patrick J.: Fluid-Dynamic Properties of Some Simple Sharp- and Blunt-Nosed Shapes at Mach Numbers from 16 to 24 in Helium Flow. NASA Memo 5-8-59L, June 1959.**

Flat plates, 5° and 10° cones were investigated. The flat-plate results are for a leading-edge Reynolds number range of 584 to 19,500 and show that the induced pressure distribution is essentially linear with the hypersonic viscous interaction parameter X within the scope of this investigation. It is also shown that the rate at which the induced pressure varies with X is a linear function of the leading-edge Reynolds number. The wedge and cone results show that as the flow-deflection angle increases, the induced-pressure effects decrease and the measured pressures approach those predicted by inviscid shock theory.

35. **Close, William H.: Hypersonic Longitudinal Trim, Stability and Control Characteristics of a Delta-Wing Configuration at High Angles of Attack. NASA TM X-240, April 1960.**

Results show that, through the use of deflected nose and trailing-edge flaps, these configurations could trim throughout the test angle-of-attack range and still maintain static longitudinal stability and control. A method is presented for predicting longitudinal stability characteristics of delta wings at high angles of attack. The effects of utilizing unported trailing-edge flaps were also investigated.

36. **Penland, Jim A.; and Armstrong, William O.: Static Longitudinal Aerodynamic Characteristics of Several Wing and Blunt-Body Shapes Applicable for Use as Reentry Configurations at a Mach Number of 6.8 and Angles of Attack Up to 90°. NASA TM X-65, October 1959.**

This investigation which was made in the Langley 11-inch hypersonic tunnel showed that wing planform had no noticeable effect upon the lift-drag characteristics at very high angles of attack and that the maximum lift coefficient was not appreciably changed with a variation in Mach number in the high hypersonic speed range. The blunt-body shapes were found to have high pressure drag which could be altered by changing the face geometry or by the addition of flaps, to be statically stable about a practical center-of-gravity location, and to be capable of being trimmed by the use of flaps to permit limited control over a range and reentry deceleration.

37. **Armstrong, William O.; Ladson, Charles L.; Baradell, Donald L.; and Blackstock, Thomas A.: Effects of Variation in Body Orientation and Wing and Body Geometry on Lift-Drag Characteristics of a Series of Wind-Body Combinations at Mach Numbers From 3 to 18. NASA TM X-73, October 1959.**

The investigation was made in the Langley 11-inch hypersonic tunnel at Mach numbers of 6.86 and 9.6 in air and 10.5, 13.0, and 17.8 in helium. Some additional data were obtained in the Langley 9-inch supersonic tunnel at a Mach number of 2.91. Lift drag, and pitching-moment data were measured for angles of attack from -3° to 16° at several Reynolds numbers. The extent to which some of the results could be predicted by use of several existing theories was also assessed. The design and calibration of the helium nozzles are also included as an appendix.

38. **Stainback, Calvin P.: An Experimental Investigation at a Mach Number of 4.95 of Flow in the Vicinity of a 90° Interior Corner Alined With the Free-Stream Velocity. NASA TN D-184, 1960.**

39. **McLellan, Charles H.; and Ladon, Charles L.: A Summary of the Aerodynamic Performance of Hypersonic Gliders. NASA TM X-237, March 1960.**

No significant favorable interference effects appear to arise from placing the body under the wing above a Mach number of about 10 and selection of a configuration should be based on other considerations such as heating or structural requirements. At lower Mach numbers, the results indicted a lift-drag ratio advantage of some flat-top configurations over the same configuration inverted to form a flat-bottomed one; however, there are other factors to be considered. Maximum lift-drag ratio results of hypersonic glider tests appear to agree well with the expected trend with Mach number established by a compilation of data from supersonic configurations tested in the Mach number range from 1.5 to 5.

40. **Close, William H.: Hypersonic Longitudinal Trim, Stability, and Control Characteristics of a Delta-Wing Configuration at High Angles of Attack. NASA TM X-240, April 1960.**

41. **Mueller, James N.; Close, William H.; and Henderson, Arthur, Jr.: An Investigation of Induced-Pressure Phenomena on Axially Symmetric, Flow-Aligned, Cyindrical Models Equipped With Different Nose Shapes at Free-Stream Mach Numbers From 15.6 to 21 in Helium. NASA TN D-373, May 1960.**

Induced-pressure-distribution studies were made in a 2-inch helium tunnel and the Langley 11-inch hypersonic tunnel on six nose shapes: a hemisphere, at 45° included-angle cone, a hemisphere modified by a 90° included-angle conical tip, and a flat configuration. Pressures were measured on the models from 0 to 50 model diameters from the nose-cylinder juncture. The effects of Reynolds numbers were determined in the range from 0.062 to 0.75×10^6 . Comparisons were made of data obtained in a conical, or Mach number gradient, nozzle with data obtained in a uniform-flow nozzle (no Mach number gradient).

42. **Bertram, Mitchel H.; and Henderson, A.: Recent Hypersonic Studies of Wings and Bodies. American Rocket Soc. Semi-Annual Meeting, Los Angeles, California, May 9-12, 1960.**

Gives information obtained at Langley at Mach numbers ranging from 7 to 21 encompassing both work in air and helium on shapes ranging from rods to delta wings. Results indicate that, in most cases, methods for making useful estimates of pressure are in hand for the simple shapes. However 3-dimensional effects and the interaction between the components considerably complicates the flow fields over delta wings at low angles of attack.

43. **Penland, Jim A.: Static Longitudinal Stability of a Missile Configuration with Various Nose Shapes and Flared Afterbodies at a Mach Number of 6.82. NASA TM X-274, June 1960.**

An investigation to determine the variation of the static longitudinal stability of a cylindrical missile configuration with four variations of nose shape, three of flare angle, and three of flare length has been carried out with average Reynolds number of 2.8×10^6 based on body length. Six-component force data were obtained for a from -5° to 15° .

44. **Bertram, Mitchel H.; Feller, William V.; and Dunavant, James C.: Flow Fields, Pressure Distributions, and Heat Transfer for Delta Wings at Hypersonic Speeds. NASA TM X-316, September 1960.**

Successful correlations of pressure on delta-planform wings are shown for both the centerline and spanwise pressure distributions over a range of sweep, Mach number angle of attack, dihedral angle. Such correlations allow the application of experimental results to Mach numbers widely different from those at which the data were obtained. Much of the heat transfer has been found to be amenable to simple approaches which take into account the flow pattern peculiar to the angle-of-attack range under consideration. The leading edge itself has been found to become a trailing edge in the sense of airflow direction at high angles of attack.

45. **Armstrong, William O.; Stainback, P. Calvin; and McLellan, Charles H.: The Aerodynamic-Force and Heat-Transfer Characteristics of Lifting Reentry Bodies. NASA TM X-352, August 1960.**

Most of the force data were obtained at a Mach number of 9.6 in air; however, several configurations were tested through a Mach number range from about 3 to 18. The basic heat-transfer characteristics were obtained from theoretical considerations. This study has served to point out some of the relative merits of several classes of these lifting-body characteristics.

46. **Henderson, Arthur, Jr.: Investigation of the Flow Over Simple Bodies at Mach Numbers of the Order of 20. NASA TN D-449, August 1960.**

It is shown that adequate means are available for calculating inviscid direct and induced pressures on simple axisymmetric bodies at zero angle of attack. The extent to which viscous effects can alter these predictions is indicated. It is also shown that inviscid induced pressure can significantly affect the stability.

47. **Baradell, Donald L.; and Bertram, Mitchel H.: The Blunt Plate in Hypersonic Flow. NASA TN D-408, October 1960.**

The sonic-wedge characteristics method is used to investigate shock shapes and surface pressure distributions on several blunt two-dimensional shapes in a hypersonic stream. These shapes include the blunt slab aligned with the stream and at an angle of attack and power profiles of the form y proportional to x^m where $0 < m < 1$. The effects of a free-stream conical-flow gradient on the pressure distribution on a blunt slab in hypersonic flow are investigated and procedures for reducing pressure data obtained in conical flows with small gradients are presented.

48. **Van Hise, Vernon: Analytical Study of Induced Pressure on Long Bodies of Revolution with Varying Nose Bluntness at Hypersonic Speeds. NASA TN D-437, 1960.**

Pressure distributions and shock shapes for a series of cylindrical afterbodies having nose fineness ratios from 0.4 to 4 have been calculated by using the method of characteristics for a perfect gas. The fluid mediums investigated were air and helium and the Mach number range was from 5 to 40. Flow parameters obtained from blast-wave theory give good correlations of blunt-nose induced pressures and shock shapes. Experimental results are found to be in good agreement with the characteristic calculations. The concept of hypersonic similitude enables good correlation of the results with respect to body shape, Mach number, and ratio of specific heats.

49. **Ware, George M.: Static Stability and Control Characteristics at Low-Subsonic Speeds of a Lenticular Reentry Configuration. NASA TM X-431, December 1960.**

The model used in the investigation was lenticular in shape (double convex in cross section and circular in planform) and had flat-plate horizontal fins with end plates. Data are presented for the basic model and for the model with several canopy designs and fin modifications.

50. **Ladson, Charles L.; and Johnston, Patrick J.: Aerodynamic Characteristics of Two Winged Reentry Vehicles at Supersonic and Hypersonic Speeds. NASA TM X-346, February 1961.**

Tests on two performance, stability, and control data are presented at Mach numbers of 1.62 and 2.91 for angles of attack up to 15° and at Mach numbers of 6.8 and 9.6 for angles of attack up to 25° .

51. **Everhart, Philip E.; and Bernot; Peter T.: Measurements of the Surface Flows, Heat Transfer, Pressure Distribution, and Longitudinal Stability of a Mercury Capsule Model at Mach Numbers of 6.9 and 9.6. NASA TM X-458, April 1961.**

Pressure-distribution and heat-transfer tests on a 0.03-scale model of the Mercury capsule reentry configuration have been made in the Langley 11-inch hypersonic tunnel at a Mach number of 9.6, a Reynolds number of 240,000 based on the model maximum diameter, and an angle-of-attack range from 0° to 15° . Schlieren and oil-flow tests were made at Mach numbers of 6.9 and 9.6 in order to study the capsule flow field. Static longitudinal stability characteristics of the escape, exit, and reentry capsule configurations at a Mach number of 9.6 are also presented.

52. Jones, Robert A.; and Galagher, J. J.: Heat Transfer and Pressure Distribution of a 60° Swept Delta Wing With Dihedral at a Mach Number of 6 and Angles of Attack From 0° to 52°. NASA TM X-544, March 1961.

53. Henderson, Arthur, Jr.; and Braswell, Dorothy O.: Charts for Conical and Two-Dimensional Oblique-Shock Flow Parameters in Helium at Mach Numbers From About 1 to 100. NASA TN D-819, June 1961.

Charts are presented for the determination of certain flow properties in the flow field and on the surface of cones and wedges at Mach numbers from about 1 to 100 in a perfect gas with a ratio of specific heats of 5/3. In addition, a table of the isentropic subsonic flow parameters is included.

54. Matting, Fred W.; Chapman, Dean R.; Nyholm, Jack R.; and Thomas, Andrew G.: Turbulent Skin Friction at High Mach Numbers and Reynolds Numbers in Air and Helium. NASA TR R-82, 1961.

55. Penland, Jim A.: Aerodynamic Force Characteristics of a Series of Lifting Cone and Cone-Cylinder Configurations at a Mach Number of 6.83 and Angles of Attack Up to 130°. NASA TN D-840, June 1961.

Force tests of a series of right circular cones having semi-vertex angles ranging from 5° to 45° and a series of right circular cone-cylinder configurations having semi-vertex angles ranging from 5° to 20° and an afterbody fineness ratio of 6 have been made in the Langley 11-inch hypersonic tunnel at a Reynolds number of $.24 \times 10^6$ per inch. An analysis of the results made use of the Newtonian theories and the exact theory.

56. Wagner, Richard D., Jr.; Pine, W. Clint; and Henderson, Arthur, Jr.: Laminar Heat-Transfer and Pressure-Distribution Studies on a Series of Reentry Nose Shapes at a Mach Number of 19.4 in Helium. NASA TN D-891, June 1961.

The pressure and heat-transfer-rate distributions on the nose shapes are compared with theoretical predictions to ascertain the limitations and validity of the theories of hypersonic speeds. Two of the nose shapes were tested with variable-length flow-separation spikes. The results obtained by previous investigators of spike-nose bodies were found to prevail at the higher Mach number of the present investigation.

57. Ware, George M.; and Shanks, Robert E.: Investigation of the Low Subsonic Stability and Control Characteristics of a Model of a Hypersonic Boost-Glide Airplane Designed for High Lift-Drag Ratios at Low Speeds. NASA TM X-534, June 1961.

The investigation included static and dynamic force tests of a model with a 70° swept wing, a teardrop-shaped fuselage, and wing-tip vertical tails. Tests were also conducted to determine the effect of various configuration modifications made in an effort to improve the low-speed stability and the lift-drag ratios.

58. **Bertram, Mitchel H.; and Henderson, A.: Recent Hypersonic Studies of Wings and Bodies. Am. Rocket Soc. Journal, vol. 31, no. 8, August 1961, pp. 1129-1139.**

Even though a great deal of theoretical and experimental information has been obtained in recent years on the flow over simple shapes in hypersonic flow, a great deal of confusion still exists on how to interpret and extrapolate the results obtained. This paper offers information recently obtained at Langley at Mach numbers ranging from 7 to 231 encompassing both work in air and helium on shapes ranging from rods to delta wings. The results indicate that in most cases methods for making useful estimates of pressure are in hand for the simple shapes. However, three-dimensional effects and the interaction between the components considerably complicates the flow fields over delta wings.

59. **Dunavant, J. C.: Investigation of Heat Transfer and Pressures on Highly Swept Flat and Dihedral Delta Wings at $M = 6.8$ and 9.6 at Angles of Attack to 90° . NASA TM X-589, September 1961.**

60. **Coe, Frank S. III; and Feller, William V.: Experimental Investigation of the Pressures, Heat Transfer, and Surface Flow Patterns Around a Blunt Half-Cone Lifting Reentry Body at Mach No. of 9.6 . NASA TM X-589, September 1961.**

Experimental measurements of surface pressure, heat-transfer coefficients, and surface oil-flow patterns are presented for a configuration consisting of a half cone with a canted flat nose and rounded edges. The results are compared with values found for blunt delta wings and with predictions by local flat-plate theory.

61. **Wagner, Richard D.: An Experimental Study of Nose Shape and Reynolds Number Effects on Induced Pressures at a Mach Number of 24 . NASA TM X-51012, August 1962. (Also available as M.S. Thesis, Va. Poly. Institute, August 1961).**

62. **Sterrett, James R.; and Emery, James C.: Experimental Separation Studies for Two-Dimensional Wedges and Curved Surfaces at Mach Numbers of 4.8 to 6.2 . NASA TN D-1014, 1962.**

63. **Henderson, Arthur, Jr.: Recent Investigations of the Aerodynamic Characteristics of General and Specific Lifting and Nonlifting Configurations at Mach of 24 in Helium Including Air-Helium Simulation Studies. Presented at the Specialists' Meeting on the High-Temperature Aspect of Hypersonic Flow, AGARD, Rhode-Saint-Genese, Belgium, April 3-6, 1962 (N62-12469).**

Recent hypersonic helium results are presented in the Mach number range of about 11 to 24 on various general and specific configurations. These include studies of induced pressures on cylindrical rods behind various nose shapes, variation of the longitudinal aerodynamic characteristics of a series of low fineness ratio bodies with systematic changes in geometry, and an experimental air-helium simulation investigation.

64. **Henderson, Arthur, Jr.: Recent Investigations of the Aerodynamic Characteristics of General and Specific Lifting and Nonlifting Configurations at Mach 24 in Helium, Including Air-Helium Simulation Studies. April 1962.**

Recent hypersonic helium results are presented in the Mach number range of about 11 to 24 on various general and specific configuration. These include studies of induced pressures on cylindrical rods behind various nose shapes, variation of the longitudinal aerodynamic characteristics of a series of low fineness ratio bodies with systematic changes in geometry, and an experimental air-helium simulation investigation.

65. **Witcofski, Robert D.; and Henderson, Arthur, Jr.: Induced Pressures on Cylindrical Rods With Various Nose Drags and Nose Shapes at Mach Numbers of 17 and 21. NASA TN D-1266, May 1962.**

66. **Ware, George M.; and Shanks, Robert E.: Investigation of the Low-Subsonic Flight Characteristics of a Model of a Reentry Configuration Having a 75° Delta Wing. NASA TM X-684, May 1962.**

The model was tested with two wing-tip vertical-tail configuration: one with a pentagonal planform and the other with a triangular planform. Flight tests were made in the Langley full-scale tunnel over an angle-of-attack range from about 13° to 45°. Static and dynamic force tests were also made over an angle-of-attack range from 0° to 45°.

67. **Shanks, Robert E.; and Ware, George M.: Investigation of the Flight Characteristics of a 1/5-Scale Model of a Dyna-Soar Glider Configuration at Low-Subsonic Speeds. NASA TM X-683, June 1962.**

68. **Dunavant, James C.: Investigation of Heat Transfer and Pressures on Highly Swept Flat and Dihedral Delta Wings at Mach Numbers of 6.9 and 9.6 and Angles of Attack to 90°. NASA TM X-688, June 1962.**

69. **Goldberg, T. J.; and Hondros, J. G.: Pressure Distributions on a Flat-Plate Delta Wing Swept 65° at a Mach Number of 5.97 at Angles of Attack from 65° to 115° and Angles of Roll from 0° to 25° at a 90° Angle of Attack. NASA TM X-702, August 1962.**

70. **Hondros, James G.: Pressure Distributions on Two-Dimensional Sharp-Leading-Edge Flat Plates With Sweep Angles of 0°, 30°, and 45° at a Mach Number of 6 and Angles of Attack From 0° to 90°. NASA TN-D1371, September 1962.**

71. **Bertram, Mitchel H.; Fetterman, David E.; and Henry, J. R.: The Aerodynamics of Hypersonic Cruising and Boost Vehicles. NASA SP-23, Aerodynamics of Space Vehicles, December 1962, pp. 23-42.**

The problems of efficient hypersonic flight of boostglide and air-breathing vehicles are reviewed for the areas of aerodynamics, stability and control, heating, and air ingestion. The application of classical hypersonic solutions to this class of vehicles is shown although these solutions cannot always be applied without extensive modification. Particular attention is given to the problems of interference between major vehicle components and the internal flow problems of air breathers.

72. **Ware, George M.: Investigation of the Low-Subsonic Aerodynamic Characteristics of a Model of a Modified Lenticular Reentry Configuration. NASA TN X-756, December 1962.**

73. **Penland, Jim A.: Theoretical and Experimental Longitudinal and Lateral Static Stability and Control Characteristics of a Series of Axisymmetric High Drag Reentry Vehicles at a Mach Number of 6.7. NASA TM X-770, March 1963.**

Six-component force tests of a series of axisymmetric, high-drag, reentry vehicles have been made at a Mach number of 6.7 and a Reynolds number of 0.23×10^6 based on diameter at angles of attack up to 360° . Vehicle designs included variations of the heat-shield edge radii; the heat-shield curvatures, and the flaps. Calculations using the modified Newtonian theory were made on the configurations for all parameters at all angles of attack. A study was made of the effect of center-of-gravity location upon trim.

74. **Fetterman, David E.; and Neal, Luther, Jr: An Analysis of the Delta-Wing Hypersonic Stability and Control Behavior at Angles of Attack Between 30° and 90° . NASA TN D-1602, March 1963.**

An experimental investigation has been conducted to determine the force, stability, and control characteristics of a 70° sweep delta wing at a Mach number of 6.83. Now incidence (provided by a forward panel 16 percent of the wing area) and a trailing-edge flap (36- percent wing area) were used as the trim devices. The results were obtained by integrating extensive pressure measurements over the windward surface of the wing and thereby making possible a determination of not only the total forces and moments produced but also the contributions of the local nose, middle, and flap components, which facilitated a dependable analysis of the various behavior encountered. The scope of measurements included angles of attack between 30° and 90° , nose incidence values of 0° , 10° , and 20° , and flap deflections from 10° to 90° , at a Reynolds number of 0.60×10^6 .

75. **Fuller, Dennis E.; Shaw, David S.; and Wassum, Donald L.: Effect of Cross-Section Shape on the Aerodynamic Characteristics of Bodies at Mach Numbers From 2.50 to 4.63. NASA TN D-1620, March 1963.**

76. **Watson, Ralph D.; and Wagner, Richard D., Jr.: Pressure Distribution at a Mach Number of 24.5 on a Symmetrical Blunt-Faced Reentry Body at Angles of Attack from 0° to 40° in Helium Including an Investigation of Afterbody Sting Effects. NASA TM X-841, May 1963.**

77. **Goldberg, Theodore J.; Ashby, George C., Jr.; and Hondros, James G.: Center-Line Pressure Distributions on Two-Dimensional Bodies with Leading-Edge Angles Greater Than That for Shock Detachment at Mach Number 6 and Angles of Attack up to 25°. NASA TN D-1793, June 1963.**

Centerline pressure distributions were obtained for two-dimensional sharp-nose parabolic arc, circular arc, and wedge bodies having a leading-edge angle greater than that for shock detachment (aerodynamically blunt bodies) at Mach number of 6 for angles of attack up to 25°. The maximum pressure coefficient was found to increase continuously from the shock-attachment value to the stagnation value behind a normal shock between leading-edge deflection angles of 42° and 51°. Only the data for contoured bodies having leading-edge angles of 66° or greater are correlated very well by the generalized Newtonian theory. However, at all angles of attack for all aerodynamically blunt bodies having curved surfaces, the agreement between the generalized Newtonian theory and the measured values of pressure coefficient was reasonably good for surface-deflection angles above 30°.

78. **Bernot, Peter T.: Pressure Distributions on Blunt Delta Wings at Angles of Attack up to 90° and Mach Number of 6.85. NASA TN D-1954, July 1963.**

The results of an investigation conducted on a family of delta wings are presented in the form of tabulated pressure coefficients, pressure-distribution plots, schlieren photographs, and oil-flow studies. The effects of wing sweep angle and angle of attack are assessed. A comparison of the experimental data with several hypersonic theories is also presented.

79. **Wagner, Richard D., Jr.; and Waston, Ralph D.: Reynolds Number Effects on the Induced Pressures of Cylindrical Bodies With Different Nose Shapes and Nose Drag Coefficients in Helium at a Mach Number of 24. NASA TR R-182, November 1963.**

Exact inviscid pressure distributions and shock shapes, obtained by the method of characteristics, are compared with experimental data. The results indicate a strong dependence of induced pressures upon Reynolds number, especially in the region immediately downstream of the nose-cylinder junction. The measured shock shapes revealed no discernible effect of Reynolds number variations.

80. **Bertram, Mitchel H.; and Everhart, Philip E.: An Experimental Study of the Pressure and Heat-Transfer Distribution on a 70 Sweep Slab Delta Wing in Hypersonic Flow. NASA TR R-153, December 1963.**

Results are presented for a study of the pressure and heat-transfer distributions and force characteristics of slat delta wings of 70° sweep at Mach numbers 6.8 and 9.6 in air and 18 in helium. The wings had cylindrical leading edges and were tested with two different noses. One was sharp in plan view, and the other was a tangent sphere with the same diameter as the cylindrical leading edge. Simple approaches to predict the heat transfer are shown to be successful if the flow pattern peculiar to the angle-of-attack range under consideration is taken into account.

- 81. Henderson, Arthur, Jr.: Recent Investigations of the Aerodynamic Characteristics of General and Specific Lifting and Nonlifting Configurations at Mach 24 in Helium, Including Air-Helium Simulation Studies. AGARDOGRAPH-68, The High Temperature Aspects of Hypersonic Flow, 1964, pp. 163-190. (Also available from New York, MacMillan Co., Oxford, Pergamon Press, Ltd. 1964, p. 163-190.**

Recent hypersonic helium results are presented in Mach 11 to 24 on various general and specific configurations. These include studies of induced pressures on cylindrical rods behind various nose shapes variation of the longitudinal aerodynamic characteristics of a series of low fineness ratio bodies with systematic changes in geometry, and an experimental air-helium simulation investigation. The induced pressure study showed viscous effects to be significant in the region near the shoulder where the blast-wave theory is known to be inapplicable and far downstream where, at lower Mach number, the blast-wave theory is adequate. The low fineness-ratio-body investigation presents a wide variety of shapes and their corresponding aerodynamic characteristics. The range of configurations examined includes shapes that may closely satisfy several plausible design criteria for planetary entry probes. The investigation consisted of force and moment studies on various general and specific aerodynamic shapes: the general shapes were tested at Mach 9.6 in air and 10.9 in helium, the specific configurations were tested at a Mach 24 in helium and compared with data on the same configurations for Hotshot tunnels in Mach 15 to 22 range.

- 82. Bertram, Mitchel H.; Fetterman, David E.; and Henderson, Arthur, Jr.: Studies Relating to the Attainment of High Lift-Drag Ratios at Hypersonic Speeds. NASA TM X-54706, 1964.**

An evaluation of results of recent investigations of slender configurations to determine factors having the most significant influence on aerodynamic efficiency is presented. The results, which were obtained at a Mach number of 6.8 in air and of 20 in helium, show the effects on maximum lift-drag ratio of viscosity, body longitudinal curvature, cross-sectional shape, and fineness ratio, wing location, planform leading-edge sweep and diameter, and volumetric efficiency. In addition, the interrelationship of these factors in determining the extent of beneficial effects from favorable interference is examined. Included also are preliminary comparisons between certain merged wing-body configurations and discrete wing-body types.

- 83. Henderson A., Jr.:Recent Investigations of the Aerodynamic Characteristics of General and Specific Lifting and Nonlifting Configurations at Mach 24 in Helium, Including Air-Helium Simulation Studies. AGARDOgraph 68, 1964, pp. 163-168.**

- 84. Wagner, Richard D., Jr. and Watson, Ralph: Induced Pressures and Shock Shapes on Blunt Cones in Hypersonic Flow. NASA TN D-2182, March 1964.**

Numerical solutions and experimental pressure distributions have been studied with the intention of evaluating proposed extensions to the case of blunt cones of the blast-wave correlation of induced pressures. The numerical inviscid solutions obtained by characteristics theory do, with some success, correlate in terms of what may be considered as modified blast-wave parameters. A secondary study of Reynolds number effects indicates that the poorer correlation of experimental data is due, in part, to viscous effects. The experimental shock shapes are also obtained and found to give a good correlation for small cone angles in terms of blast-wave-type parameters.

- 85. Witcofski, Robert D.; and Woods, William C.: Static Stability Characteristics of Blunt Low-Fineness-Ratio Bodies of Revolution at a Mach Number of 24.5 in Helium. NASA TN D-2282, May 1964.**

- 86. Penland, Jim A.: A Study of the Stability and Location of the Center of Pressure on Sharp, Right Circular Cones at Hypersonic Speeds. NASA TN D-2283, May 1964.**

Analysis of experimental data obtained on various sharp, right circular cone models at a Mach number of 6.8 and a Reynolds number of 0.22×10^6 per inch indicates that the Newtonian theory gives good predictions of the longitudinal stability and performance parameters through a 30° angle-of-attack range for cones with semivertex angles less than 40° . The location of the center of pressure was found to be fixed for a given cone angle and independent of angle-of-attack change. A decrease in Reynolds number by a factor of 5 made no change in the longitudinal characteristics of a 5° semivertex cone except an increase of drag due to skin friction. The calculated longitudinal characteristics are presented for cones with semivertex angles of 2.5° to 50° in Newtonian flow for angles of attack from 0° to 180° with both base area and planform area reference.

- 87. Blackstock, Thomas A.; and Ladson, Charles L.: Comparison of the Hypersonic Aerodynamic Characteristics of Some Simple Winged Shapes in Air and Helium. NASA TN D-2328, June 1964.**

An investigation was made in the Langley 11-inch hypersonic tunnel at Mach numbers of 6.8 and 9.6 in air and 10.9 in helium to determine the force and moment characteristics of a series of wings. This investigation was made to study the simulation of high Mach number aerodynamics in air by the use of helium as a test medium. The wings tested were of both square and delta planform and included both sharp and blunt leading edges. Also the effects of a vertical forward-facing step on the characteristics of the square wing were investigated. The angle-of-attack range for the tests was from 0° to 25° . Analysis of the results indicated that lift and drag coefficients could be predicted over the range of test Mach numbers and for the test media used. Drag coefficients could not be as adequately predicted as lift coefficients due to viscous effects. Good simulation was obtained for normal-force coefficients, however. The forward-facing step on the square wing produced a large increase in lift and pitching-moment coefficients and may be used as a pitch control device. Two methods of correlating normal-force coefficients are also presented.

88. **Neal, Luther Jr.; and Fetterman, David E.: Control-Surface Interaction Effects on Delta-Wing Windward Pressures at a Mach Number of 6.83 at High Angles of Attack. NASA TN D-2332, June 1964.**

An experimental investigation has been conducted to determine the effects of nose incidence and trailing-edge-flap deflections on the windward pressure distribution over a 70° sweep delta wing at a Mach number of 6.83. The study was conducted over an angle-of-attack range from 30° to 90° with nose incidence of 0° to 20° and flap deflections of 10° to -90° at a Reynolds number of 0.6×10^6 , based on mean aerodynamic chord.

89. **Ware, George M.: Effect of Fin Arrangements on Aerodynamic Characteristics of a Thick 74° Delta Wing Manned Lifting Entry Vehicle at Low-Subsonic Speeds. NASA TM X-1020, June 1964.**

90. **Beckwith, Ivan E.: Experimental Investigation of Heat Transfer and Pressures on a Swept Cylinder in the Vicinity of Its Intersection with a Wedge and Flat Plate at Mach Number 4.15 and High Reynolds Numbers. NASA TN D-2020, July 1964.**

Heat-transfer rates and pressures have been measured on a cylinder at sweep angles of 20° and 60° in the interference region between the cylinder and an 8 half-angle wedge. Data were also obtained with the cylinder mounted on a flat plate at a sweep angle of 20°. Comparison of the heat-transfer data with theory indicated that the flow was fully turbulent. The result shows that at 20° sweep, the large increases in pressure and heating that occur near the intersection can be attributed to local flow conditions and separation on the wedge. At 60° sweep the increases are much less and are caused entirely by local wedge flow. There was no effect of the wedge shock impingement on heating rates or pressures on the cylinder.

91. **Staylor, W. Frank; and Goldberg, Theodore J.: Afterbody Pressures on Two-Dimensional Boattailed Bodies Having Turbulent Boundary Layers at Mach 5.98. NASA TN D-2350, July 1964.**

An experimental investigation has been conducted on a series of two-dimensional afterbodies to determine the effects of boattailing and angle of attack upon base and boattail pressures. Afterbodies with boattail angles for 0° to 18° at angles of attack up to 14° were investigated at Reynolds numbers sufficient to cause fully turbulent boundary layers to exist ahead of the afterbodies. A correlation of base-pressure data from previous investigations was made with that from the present study and a semiempirical method is presented which gives a reasonable estimate of the base pressures between the Mach numbers of 1.4 to 6.0. The empirical estimation of boattail pressures made possible predictions of afterbody and drag.

92. **Putnam, Lawrence E.; and Trescot, Charles D., Jr.: Hypersonic Aerodynamic Characteristics of Plain and Ported Elevon Controls On A 75° Swept Modified Delta-Wing Configuration. NASA TM X-987, July 1964.**

An investigation has been made in the Langley 15-inch hypersonic flow apparatus to measure the relative effectiveness of plain and ported elevons on a 75° swept, clipped-tip, delta-wing configuration. The effect of the vertical location of the wing (fuselage on the top or bottom of the wing) on elevon effectiveness was also investigated. The results were obtained at a Mach number of 10.03 at angles of attack from about -4° to 21°. The plain and ported elevons (on left wing only) were deflected in 15° increments from -45° to 45° on the low-wing configuration and from 0° to 45° on the high-wing configuration. The Reynolds number, based on the wing mean aerodynamic chord, varied from about 0.51×10^6 to about 0.80×10^6 .

93. **Becker, J. V.: Studies of High Lift/Drag Hypersonic Configurations. TMX-51776. (Proceedings, 4th ICAS Congress Palais de l'UNESCO, Paris, France) August 1964.**

94. **Spencer, Bernard, Jr.; Phillips, W. Pelham; and Fournier, Roger H.: Supersonic Aerodynamic Characteristics of a Series of Bodies Having Variations in Fineness Ratio and Cross-Sectional Ellipticity. NASA TN D-2389, August 1964.**

An investigation was conducted in the Langley Unitary Plan wind tunnel to determine the effects of cross-sectional ellipticity and fineness ratio on the longitudinal and lateral aerodynamic characteristics of a series of 2/3-power low-wave-drag bodies at Mach 1.50 to 2.86. Reynolds number per foot was held constant at 2.75×10^6 for all Mach numbers. The angle of attack range was from approximately -50 to 29 at 0° and 5° of sideslip. Increasing body major-to-minor axis ratio with the major axis horizontal, resulted in large increases in the lift, drag, and pitching-moment coefficients and lift-drag ratios at positive angles-of-attack and at all test Mach numbers. The successive increases in major-to-minor axis ratio, with the major axis horizontal, resulted in gains in lift-curve slope for bodies of the same fineness ratio when the coefficients were based on the respective body projected planform areas; generally, only slight changes in the minimum-drag characteristics of the bodies were noted. A rearward shift in the body center-of-pressure location is generally indicated when the horizontal-to-vertical axis ratio is increased from 0.50 to 2.00. Little or no effects of increasing Mach number on the center-of-pressure location of a given body were shown.

95. **Stainback, P. Calvin: Heat-Transfer Measurements at a Mach Number of 8 in the Vicinity of a 90° Interior Corner Alined With the Free-Stream Velocity. NASA TN D-2417, August 1964.**

An experimental investigation was conducted to determine the influence of the interaction of the flow field with boundary layer on the heat transfer in the vicinity of an interior corner. The investigation was made at a nominal Mach number of 8 and the nominal unit Reynolds number per foot was varied from 0.23×10^6 to 10.07×10^6 . The results indicated a low heating region very near the corner, which was associated with the interaction of the boundary layers in the vicinity of the corner. A high heating region was found far from the corner in comparison with the boundary-layer thickness and between the corner and the shock.

96. **Suddath, Jerrold H., and Oehman, Waldo I.: Minimum Drag Bodies with Cross-Sectional Ellipticity. NASA TN D-2432, September 1964.**

Newtonian flow theory and the calculus of variations were used to study minimum drag shapes for bodies with cross-sectional ellipticity and with given length and base height, length and volume, base height and volume, and base height and surface area. Numerical examples for given length and volume are presented to assess the effect of cross-sectional ellipticity on body shape and pressure drag coefficient.

97. **Arrington, James P.; Joiner, Roy C. Jr.; and Henderson, Arthur Jr.: Longitudinal Characteristics of Several Configurations at Hypersonic Mach Numbers in Conical and Contoured Nozzles. NASA TN D-2489, September 1964.**

Pointed cones, a slender cone-cylinder, a winged ellipsoid, and several cylindrical configurations have been tested in helium in a contoured nozzle at a Mach number of 20.5 and in a conical nozzle at a Mach number of about 24. The effects of flow in a conical nozzle on the aerodynamic characteristics of pointed slender bodies can be appreciable, and the extent can vary with model geometry and position. The Mach number at the model, rather than an average over the entire core, should be used in reducing the data obtained in hypersonic nozzles; otherwise, large discrepancies may occur in the aerodynamic results.

98. **Spencer, Bernard, Jr.; and Phillips, W. Pelham: Transonic Aerodynamic Characteristics of a Series of Bodies Having Variations in Fineness Ratio and Cross-Sectional Ellipticity. NASA D-2622, 1965.**

An investigation has been made in the Langley high-speed 7- by 10-foot tunnel to determine the longitudinal and lateral aerodynamic characteristics at transonic speeds of a series of low-wave-drag lifting bodies having variations in fineness ratio and cross-sectional ellipticity.

99. **Witcofski, Robert D; and Marcum, Don C., Jr.: Wing Sweep and Blunting Effects on Delta Planforms at $M = 20$. AIAA Journal, vol. 3, no. 3, March 1965, p. 547-548.**

This paper is a brief resume of the preliminary results obtained from an investigation recently conducted at a Mach number of 20 in helium flow to determine the effects of leading edge blunting and sweep angle on the static longitudinal stability characteristics of basic delta planforms, with particular emphasis on maximum lift-drag ratio. For the purpose of brevity, only summary plots are presented. The experimental results were obtained in the Langley 22-in. helium tunnel utilizing a contoured nozzle to obtain a uniform freestream Mach number of 20.3. Stagnation temperature for the tests was approximately 80°F. Reynolds number, based on chord length, ranged from 1.5×10^6 to 5.5×10^6 , but for wings having thickness ratios less than 0.034, a constant Reynolds number of $.37 \times 10^6$ was maintained in order to minimize any possible Reynolds number effects.

100. **Whitehead, Allen H.; and Dunavant, James C.: A Study of Pressure and Heat Transfer Over an 80 Sweep Slab Delta Wing in Hypersonic Flow. NASA TN D-2708, March 1965.**

- 101. Johnston, Patrick J.; Snyder, Curtis D.; and Witcofski, Robert D.: Maximum Lift-Drag Ratios of Delta-Wing Half-Cone Combination at a Mach Number of 20 in Helium. NASA TN D-2762, May 1965.**

Maximum lift-drag ratios of a family of delta-wing-half-cone combinations have been determined experimentally at a Mach number of 20 in helium. Reynolds numbers based on overall length varied from 2.75×10^6 to 4.35×10^6 . The semiapex angles of the delta wing ranged from 9° to 25° and the apex angles of the half-cone body ranged from 3° to 9° . Performance measurements were made with the bodies situated both above and beneath the wing.

- 102. Bertram, Mitchel H.; and Neal, Luther Jr.: Recent Experiments in Hypersonic Turbulent Boundary Layers. NASA TM X-56335, 1956. (Presented at the AGARD Specialists Meeting on Recent Developments in Boundary-Layer Research (Naples, Italy). May 10-14, 1965.**

Considering the numerous and often differing theories available for determining the behavior of compressible turbulent boundary layers a choice of prediction method is determined by the available experimental data. The paper considers this problem in the light of the results of recent experiments in a number of different facilities and from flight. These data for skin friction and heat transfer are generally from sharp flat plate and pointed cone models and are available for very cold walls up to Mach numbers of approximately 9. Mach number and wall-temperature effects from these experiments are compared to the more recent prediction methods. In addition, an evaluation of the maximum heat transfer and shear stress to be encountered on smooth flat plates and cones is made. Since boundary-layer structure is an important consideration, boundary-layer velocity and temperature profiles obtained on hollow cylinders and nozzle walls are considered.

- 103. Cary, Aubrey M., Jr.; and Ashby, George D., Jr.: A Parametric Study of the Aerodynamic Characteristics of Nose-Cylinder Flare Bodies at a Mach Number of 6.0. NASA TN D-2584, June 1965.**

- 104. Henderson, Arthur, Jr.; Watson, Ralph D.; and Wagner, Richard D., Jr.: Recent Fluid Dynamic Studies at $M = 41$ in Helium. AIAA Paper 65-399, July 1965.**

Reentry at Mach numbers on the order of 40 and higher is anticipated for future reentry vehicles. Under certain mission requirements, radiative heating loads at these high velocities may necessitate the use of relatively slender configurations for which induced pressure effects may be a controlling factor in determining aerodynamic characteristics. Recent modifications to the Langley 22-Inch Helium Tunnel have nearly doubled its previous Mach number range. The extended capability of this tunnel, coupled with previously published information from various sources, has been utilized to examine the ideal fluid dynamics of flows over several classes of general shapes over wide Mach number ranges. Viscous and bluntness induced pressures have been examined on flat plates, wedges, cylinders, and cones. Nose drag coefficient, Reynolds number, and Mach number are variables, with primary emphasis on correlation over large Mach number ranges, the extremes being $M = 7$ to $M = 41$.

105. **Fetterman, David E.; Henderson, Arthur Jr.; Bertram, Mitchel H., and Johnston, Patrick J.: Studies Relating to the Attainment of High Lift-Drag Ratios at Hypersonic Speeds. NASA TN D-2956, August 1965.**

An evaluation of results of recent investigations of slender configurations to determine factors having the most significant influence on aerodynamic efficiency is presented. The results, which were obtained at a Mach number of 6.8 in air and of 20 in helium, show the effects on maximum lift-drag ratio of viscosity, body longitudinal curvature, cross-sectional shape, and fineness ratio, wing location, planform leading-edge sweep and diameter, and volumetric efficiency. In addition, the interrelationship of these factors in determining the extent of beneficial effects from favorable interference is examined. Included also are preliminary comparisons between certain merged wing-body configurations and discrete wing-body types.

106. **Penland, Jim A.: Maximum Lift-Drag Ratio Characteristics at Rectangular and Delta Wings at Mach 6.9. NASA TN D-2925, August 1965.**

A theoretical and experimental study of a variety of rectangular and delta planform wings at a Mach number of 6.9 and a range of root-chord Reynolds numbers from 0.35×10^6 to 4.1×10^6 has been made. This study shows that good predictions of $(L/D)_{\max}$ are possible on rectangular wings but that the predicted $(L/D)_{\max}$ for delta wings is approximately 10 percent higher than that for experiment.

107. **Fetterman, David E.: Favorable Interference Effects on Maximum Lift-Drag Ratios of Half-Cone Delta-Wing Configurations at Mach 6.86. NASA TN D-2942, August 1965.**

The characteristics of half-cone delta-wing configurations is investigated under predominantly laminar boundary-layer conditions at a Mach number of 6.86 and Reynolds number based on model length of 1.43×10^6 to determine the availability of favorable interference effects for improving the maximum lift-drag ratio. Simple modification to the half-cone body to provide more volume and better volume distribution are also considered. Approximate solutions for the characteristics of half-cone winged configurations are included in the appendixes.

108. **Ladson, Charles H.: Aerodynamic Characteristics of a Manned Lifting Entry Vehicle with Modified Tip Fins at Mach 6.8. NASA TM X-1158, October 1965.**

The effects of modifications of the tip fins and center fin on the aerodynamic characteristics of a model of a manned lifting entry vehicle with negative camber, a flat bottom, a blunt leading edge, and a delta planform (designated HL-10) have been determined at a Mach number of 6.8. The configuration with modified tip and center fins was directionally and laterally stable throughout the test angle-of-attack range. The maximum trimmed lift-drag ratio was about 1.14. Roll control effectiveness increased with increasing angle of attack and with increasing positive eleven deflection angle. The yawing moment due to roll control was very small.

109. **Arrington, James, P.; and Henderson, Arthur, Jr.: Effects of Reynolds Number on the Stability of a Blunt-Nosed Flare Cylinder from Mach 17.6 to 22.2 in Helium. NASA TM X-1166, October 1965.**
110. **Spencer, Bernard Jr.: Transonic Aerodynamic Characteristics of a Series of Related Bodies With Cross-Sectional Ellipticity. NASA TN D-3203, 1966.**
111. **Small, W. J.; and Bertram, M. H.: Effect of Geometric Modifications on the Maximum Lift-Drag Ratios of Slender Wing-Body Configurations at Hypersonic Speeds. NASA TN D-3276, March 1966.**

An investigation was conducted at Mach numbers of 6.8 and 9.6 in the Langley 11-inch hypersonic tunnel to determine the effects of geometric modifications and Reynolds number on the maximum lift-drag ratio of a series of slender wing-body configurations derived from high-maximum-lift-drag-ratio arrow-wing and delta-wing-body combinations. Most of the tests were conducted with models having 77.5° and 80.0° swept wings and 5° cone bodies. The basic configurations were altered by clipping the wing tips, removing portions of the wing-root areas, and adding a small half-cylindrical afterbody to several configurations. These modifications were intended to increase the volume--planform-area ratio of the vehicle with little sacrifice of the maximum lift-drag ratio much as would be expected from volume-area considerations; however, moderate amounts of the wing tip could be removed with little decrease of the maximum lift-drag ratio. For the basic models showing favorable flow interference benefits at a Mach number of 6.8 when the body was located beneath the wing, the favorable interference continued to occur throughout the series of geometric modifications.

112. **Neal, Luther, Jr: A Study of the Pressure, Heat Transfer, and Skin Friction on Sharp and Blunt Flat Plates at Mach 6.8. NASA TN D-3312, April 1966.**

A study of the pressure, heat-transfer, and skin-friction distributions over a flat plate with both sharp and blunt leading edges has been conducted at a nominal free-stream Mach number of 6.8, over a range of free-stream Reynolds number per centimeter from about 0.02×10^6 to 0.16×10^6 , and at a wall-to-stagnation temperature ratio of about 0.5. The model consisted of a 25.4-cm-wide by 61-cm-long flat plate equipped with interchangeable leading edges having thickness of 0.0025, 0.25, and 1.27 cm. The investigation covered all three flow regimes - laminar, transitional, and turbulent.

113. **Braslow, Albert L.: A Review of Factors Affecting Boundary-Layer Transition. NASA TN D-3384, 1966.**

A brief review is made of the current state of the art of boundary-layer transition. Discussed, in various degrees of detail, are experimentally determined effects on transition of pressure gradients, surface to free-stream temperature ratio, free-stream Mach number, free-stream turbulence, noise, two- and three-dimensional-type surface roughness, and laminar boundary-layer control through suction. Certain aspects of the theoretical approach to transition are discussed and some comparisons with experiment are made. The review is intended primarily for the engineer or scientist desiring a general understanding of boundary-layer transition phenomena rather than for the active researcher in the field of fluid mechanics. Some needs for further research are indicated.

114. **Nagel, A. L.; Fitzsimmons, H. D.; and Doyle, L. B.: Analysis of Hypersonic Pressure and Heat Transfer Tests on Delta Wings With Laminar and Turbulent Boundary Layers. NASA CR-535, 1966.**

115. **Fetterman, D. E.; McLellan, C. H.; Jackson, L. R.; Henry, B. Z., Jr.; and Henry, J. R.: A Review of Hypersonic Cruise Vehicles. NASA SP-124, Conference on Aircraft Aerodynamics, May 1966, pp. 523-573.**

In an attempt to provide a focus for future aerodynamic programs in the development of hydrogen-fueled hypersonic cruise vehicles, represent status of the structural, propulsive, and aerodynamic research is examined to extract the presently known factors that significantly affect vehicle definition. Existing wing and body structural concepts and cryogenic-tankage thermal-protection systems are illustrated, possible inlet-engine arrangements are discussed, and the status of important local aerodynamic heating areas is briefly reviewed. In addition, uncertain areas which require further fundamental research and obstacles which hinder development are also pointed out.

116. **Whitehead, Allen H., Jr.: Effect of Body Cross Section and Width-Height Ratio on Performance of Bodies and Delta-Wing--Body-Combinations at Mach 6.9. NASA TN D-2886, June 1966.**

An investigation of the aerodynamic characteristics of bodies and wing-body combinations with triangular, rectangular, and elliptical body cross-sectional shapes and with body width-height ratios of 2 and 3 was conducted at a free-stream Mach number of 6.9 and a Reynolds number based on length of 1.4×10^6 . The two delta wings tested in combination with these bodies had leading-edge sweep angles of 70° and 75° . All configurations were tested in both flat-top and flat-bottom orientations.

117. **Witcofski, Robert D.; and Marcum, Donald C., Jr.: Effect of Thickness and Sweep Angle on the Longitudinal Aerodynamic Characteristics of Slab Delta Planforms at a Mach Number of 20. NASA TN D-3459, June 1966.**

118. **Penland, Jim A.: Effect of Body Nose Blunting, Control Planform Area , and Leading Edge Bluntness on Aerodynamic Control of a 5 Semivertex Cone at Mach Number 6.9. NASA TN D-3460, June 1966.**

An experimental investigation has been carried out to determine the longitudinal, lateral, and directional control characteristics of a conical configuration with a 5° semi-vertex angle and cruciform delta planform control surfaces. The investigation was conducted at a Mach number of 6.9 and a Reynolds number of 2.8×10^6 based on model length. The configuration variables include nose bluntness, and control planform area.

119. **Fournier, Roger, H.; Spencer, Bernard Jr.; and Corlett, William A.: Supersonic Aerodynamic Characteristics of a Series of Related Bodies with Cross-Sectional Ellipticity. NASA TN D-3539, 1966.**

A wind tunnel investigation is presented for determining the supersonic aerodynamic characteristics of a series of power-law bodies and of a theoretical hypersonic minimum-wave-drag body of equal length and equal volume. The Mach number range was 1.50 to 4.63, and the angle of attack was varied from approximately -4° to 28° at 0° of sidelip. Also included in the study are the effects of altering cross-sectional ellipticity for a given body while maintaining a constant longitudinal distribution of cross-sectional area. Results indicate that increasing the power-body exponent for a given value of ellipticity results in increases in the lift-curve slope at low angles of attack. For all configurations, increasing the Mach number results in large reductions in minimum drag and large increases in the maximum lift-drag ratio. A comparison of the longitudinal aerodynamic characteristics of the theoretical hypersonic minimum-wave-drag body with the series of power-law bodies indicates that, for all values of ellipticity. The lift characteristics of this body generally fall in the range noted for the power-law bodies with exponents of 0.50 and 0.66.

120. **Bernot, Peter T.; and Jordan, Bruce C.: Static Stability Characteristics of Cone and Half-Cone--Pyramid Configurations at Mach 6.83. NASA TN D-3544, August 1966.**

The longitudinal and lateral stability characteristics of a 5° semivertex angle cone and a D-body consisting of a half-cone of 5° semivertex angle with a rectangular pyramid were determined at a Mach number of 6.83 and a Reynolds number of 1.45×10^6 . These bodies were also tested in conjunction with delta wing swept back 83.3°. The D-body configurations were tested in the flat-bottom and flat-top orientations. Effects of Reynolds number variation on the cone model were also investigated.

121. **Spencer, Bernard Jr.; and Fox, Charles H., Jr.: Hypersonic Aerodynamic Performance of Minimum Wave-Drag Bodies. NASA TR R-250, November 1966.**

An experimental investigation has been made at a Mach number of 10.03 to determine the longitudinal aerodynamic characteristics of a series of power-law bodies having values of the exponent of 0.25, 0.50, 0.66, 0.75, and 1.00, and a theoretical hypersonic minimum-wave-drag body. The bodies had the same length and volume, and the theoretical body shape was determined under the geometric constraints of prescribed length and volume. For each body, cross-sectional shape was altered from circular to elliptic, while maintaining a constant longitudinal distribution of cross-sectional area. Results of the investigation may be summarized in the following observations.

122. **Bertram, M. H.; Weinstein, L. M.; Cary, A. M. Jr.; and Arrington, J. P.: Effect of Two-Dimensional Multiple-Wave Distortions on the Heat Transfer to a Wall in Hypersonic Flow. AIAA Paper 67-164, January 1967.**

Extensive tests are reported on the heating distribution on a plate in which a series of shallow protruding waves were embedded. Peak laminar heating on the first wave empirically correlated with results from a previous investigation of single waves. The general flow configuration at the first wave was, in part, amenable to analysis by simple laminar separation concepts even when the boundary-layer thickness was greater than the wave height. The peak heating on succeeding waves in the train was essentially that predicted by assuming that each wave was independent of the other waves and applying the first wave results. With a transitional or turbulent boundary layer vortices were generally found in the separated flow areas. Turbulent peak heating correlated in much the same manner as laminar results, but with a different Mach number effect. Peak heating on single waves in turbulent attached flow could be predicted, but on waves in turbulent attached flow could be predicted, but on waves which caused flow separation, peak heating was significantly high than predictions. Wave sweep of as much as 70° caused little change in laminar heat transfer. In a transitional-turbulent case, wave sweep gave a significant reduction in peak heating, but the increased heating was still substantial.

123. **Maddalon, Dal V.; and Henderson, Arthur Jr.: Boundary Layer Transition at Hypersonic Mach Numbers. AIAA Paper 67-130, January 1967.**

An experimental investigation to locate the beginning of transition from laminar to turbulent boundary layer flow has been conducted on sharp, smooth cones having semi-apex angles of 2.87° , 5° , and 10° with a uniform free stream Mach number of about 20 in the Langley 22-inch Helium Tunnel. Local Mach number at the boundary layer edge was thus varied from 7.4 to 16.6. The data indicate that local transition Reynolds number increases very rapidly with hypersonic local Mach number. Techniques used to detect onset of transition included surface pitot tube, drag force, boundary layer pitot-pressure surveys, schlieren photographs, and hot-film anemometer measurements. Displacement thickness, momentum thickness, and velocity ratio profiles were determined for laminar, transitional, and turbulent hypersonic boundary layers. A hot-film anemometer survey of the model boundary layers showed disturbances originating within the boundary layer at much lower Reynolds numbers than the Reynolds number at which transition is felt at the model surface. In addition, source flow effects on transition Reynolds number were examined at a local Mach number of 15.8.

124. **Maddalon, Dav. V.; Rogallo, Robert S.; and Henderson, Arthur, Jr.: Transition Measurements at Hypersonic Mach Numbers. AIAA Journal, vol. 5, no. 3, March 1967, pp. 540-591.**

This note presents recent results of some boundary-layer transition studies conducted in the contoured nozzle of the Langley 22-inch helium tunnel, in which the Reynolds number required for the beginning of transition was determined by measuring the model total axial force coefficient. Additional supporting data were obtained by means of pitot pressure surveys through the model boundary layer. This investigation (conducted on a 10° half-angle cone of 6-in. base diameter) is part of a larger program currently underway at the 22-in. facility, in which transitional boundary layers are being studied at local Mach numbers from 4 to 16.

125. **Dunavant, J. C.; and Stone, H. W.: Effect of Roughness on Heat Transfer To Hemisphere Cylinders at Mach Numbers 10.4 and 11.4. NASA TN D-3871, March 1967.**

126. **Weinstein, Leonard, M.; and Neal, Luther, Jr.: Hypersonic Performance of Several Basic and Modified Diamond-Cross-Section Delta-Wing Configurations. NASA TN D-3894, April 1967.**

An investigation of the hypersonic performance of several delta wings with diamond cross sections was conducted in the Langley 11-inch hypersonic tunnel, and the effects of modifications to some of these wings were determined. The modifications were made in order to remove regions of high drag and to make the configurations more compact. The unmodified models had leading-edge sweep angles ranging from 60° to 82° and ratios of base thickness to root chord from about 0.06 to 0.21. These models were tested at free-stream Reynolds numbers based on root chord from 0.3×10^6 to 3.2×10^6 .

127. **Bertram, M. H.; and Cary A. M., Jr.: Some Topics in Hypersonic Turbulent Boundary Layer, Heat Transfer, and Skin Friction. NASA SP-148, Hypersonic Aircraft Technology Conference, May 1967. (CONFIDENTIAL)**
128. **Bertram, M. H.; and Beckwith, I. E.: Boundary Layer Transition Investigations. Presented to the Aerospace Conference Boundary Layer Transition Group, San Bernadino, California, July 1967. (CONFIDENTIAL)**

A survey of the NASA-Langley Research Center programs involving both natural and forced boundary-layer transition has been made. Included are completed, current, and planned studies such as the upcoming study of the wind tunnel noise problem and studies of the effectiveness of boundary-layer trips. Investigations of transition on smooth and ablating bodies in both wind tunnels and flight are underway. In a study of correlation of the initial transition on cones, parameters were found that correlated a large body of data within a factor of 2.

129. **Bertram, M. H.; Weinstein, L. M.; Cary, A. M., Jr.; and Arrington, J. P.: Heat Transfer to a Wavy Wall in Hypersonic Flow. AIAA Journal, vol. 5, no. 10, October 1967, pp. 1760-1767.**

Extensive tests are reported, on the heating distribution on a plate, in which a series of shallow protruding two-dimensional waves were embedded. Peak laminar heating on the first wave empirically correlated with results from a previous investigation of single waves. The general flow configuration at the first wave was, in part, amenable to analysis by simple laminar separation concepts even when the boundary-layer thickness was greater than the wave height. The peak heating on succeeding waves in the train was essentially that predicted by assuming that each wave was independent of the other waves and by applying the first wave results. With a transitional or turbulent boundary layer, vortices were generally found in the separated flow areas. Turbulent peak heating correlated in much the same manner as laminar results, but with a different Mach number effect. Wave sweep of as much as 70° caused little change in laminar heat transfer. In a transitional-turbulent case, wave sweep gave little reduction in peak heating; however, increasing the chord of the swept wave did reduce peak heating in about the proportion predicted by theory for the unswept wave.

- 130. Spencer, Bernard Jr.: Hypersonic Aerodynamic Characteristics of Minimum Wave Drag Bodies Having Variations in Cross-Sectional Shape. NASA TN D-4079, September 1967.**

An investigation has been made at hypersonic speeds of a series of bodies having variations in cross-sectional shape and chamber. The longitudinal distribution of cross-sectional area for each body conformed to the theoretical shape required to minimize the zero-lift hypersonic pressure drag of circular or elliptic bodies under the geometric constraints of given length and volume. Each body tested had constant planform area, base area, and span; the only variables were cross-sectional shape, camber, and the resultant small wetted-area changes. Cross sections tested included semicircular, elliptic, triangular trapezoidal, and rectangular shapes. Results indicated that changing cross-sectional shape with either positive or negative camber had essentially no effect on the minimum-drag characteristics of any configuration tested.

- 131. Stivers, Louis S., Jr.; and Spencer, Bernard, Jr.: Studies of Optimum Body Shapes at Hypersonic Speeds. NASA TN D-4191, October 1967.**

The present study was directed toward the questions that arise in the application of optimum bodies to the design of hypersonic cruise aircraft. The considerations were divided into two parts. The first involved the calculated minimum-drag characteristics of four families of slender bodies for Mach numbers from 2 to 12. The second concerned the experimental evaluation of the effects of body cross-sectional shape on the aerodynamic performance of bodies at a Mach number of 10. The constraints in each study were body length and volume, although the constant values are different in each part of the study.

- 132. Keyes, J. W.; and Ashby, G. D., Jr.: Calculated and Experimental Hinge Moments on a Trailing-Edge Flap of 75° Swept Delta Wing at Mach 6. NASA TN D-4268, December 1967.**

The hinge moment characteristics of a trailing-edge flap on a 75° swept delta wing have been investigated at a Mach number of 6.0 and a Reynolds number of 4.03×10^6 (based on the root chord of the delta wing). This investigation was conducted at angles of attack from 30° to 90° and flap-deflection angles from 0° to 30°. The boundary layer in the area of the wing-flap junction was considered to be transitional for angles of attack up to approximately 60° and laminar for angles of attack from 60° to 90°. Both a sharp and blunt leading-edge wing and two flap aspect ratios were used in the investigation. Although the flow on the wing and flap was found to be complex, a meaningful analysis of the data can be made for the nonseparated case if the local flow regime over the wing and flap is properly classified (subsonic or supersonic) for each angle of attack and flap-deflection angle. The slope of the hinge moment curves changed at approximately the same angle of attack that the flow regimes were predicted to change when tangent-come and oblique shock theories were used. Boundary layer separation over the flap was not extensive for flap deflections less than 20°. The type of boundary layer could have a strong influence on the hinge moments for flap deflection angles large enough to cause boundary layer separation.

133. **Spencer, Bernard Jr.; and Stivers, Louis S., Jr.: Studies of Optimum Body Shapes at Hypersonic Speeds. Presented at NASA Conference on Hypersonic Aircraft Technology, Ames Research Center, May 16-18, 1967.**

134. **Blackstock, Thomas A.; and Everhart, Philip E.: Experimental Pressure Distributions and Aerodynamic Characteristics of a Low-Fineness-Ratio Cylinder at a Mach Number of 10.5 and Angles of Attack From 0° to 90°. NASA TM X-1487, December 1967.**

An investigation was made in the Langley continuous-flow hypersonic tunnel at a Mach number of 10.5 to measure the surface pressures and the force and moment characteristics of a fuel capsule designed for use on the Nimbus B weather satellite. The capsule, a fineness-ratio-2-cylinder with a recessed face, was tested at angles of attack from 0° to 90° at a Reynolds number of 0.8×10^6 based on free-stream conditions and cylinder length. At angles of attack from 37° to 90°, forces and moments were also measured on a corresponding cylinder with a flat face.

135. **Henderson, A., Jr.; Johnston, P. J.; Molloy, J. K.; and Woods, W. C.: Recent Studies of Factors Affecting High Lift-Drag Ratios at Mach 19 in Helium. Journal of Spacecraft and Rockets, AIAA, vol. 5, January 1968, pp. 90-95. (Also available as AIAA Paper 67-138, January 1967.)**

Factors affecting performance of three vehicles designed for $(L/D)_{\max}$ of three and two at $M = 19$, $R = 3.2 \times 10^6$ have been examined in the Langley 22-inch helium tunnel. Two of the vehicles were initially flat-top and flat-bottom orientations of the same elementary shape. The relative degradation in performance of these two vehicles as each was made trimmable at $(L/D)_{\max}$ and stable about all three axes, with realistic center-of-gravity positions, is examined. Prior to satisfying trim and stability requirements the $(L/D)_{\max}$ of the flat-bottom and flat-top shapes were 3.9 and 3.1, respectively. After satisfying the stability requirements both configurations achieved a trimmed $(L/D)_{\max}$ of about 3. Thus, with the imposition of the foregoing practical design constraints the superiority of one orientation over the other will depend on considerations other than aerodynamic performance.

136. **Creel, Theodore R., Jr.; Miller, Charles G., III; Maddalon, Dal V.; and Watson, Ralph D.: Pressure Distributions and Aerodynamic Characteristics of 90° and 15° Blunted Cones in Nitrogen and Helium at Mach 20. NASA TN D-4314, March 1968.**

An experimental investigation to determine the pressure distributions and force and moment characteristics on a 9° semiapex angle spherically blunted cone was conducted in nitrogen in the Langley hotshot tunnel. Pressure distributions were obtained in the Langley 22-inch helium tunnel on 9° semiapex-angle spherically-blunted cone. The results of this investigation indicated that pressure distributions on the 9° and 15° semiapex-angle spherically-blunted cones at zero angle of attack were unpredicted by ideal gas inviscid theory, but a theory accounting for viscous effects predicted the pressure distribution on the 9° spherically-blunted cone in nitrogen.

137. **Johnson, Charles B.: Pressure and Flow-Field Study at Mach 8 of Flow Separation on a Flat Plate With Deflected Trailing-Edge Flap. NASA TN D-4038, March 1968.**

- 138. Anders, John B.; and Edwards, Clyde L. W.: A Real-Gas Study of Low Density Wedge-Induced Laminar Separation on a Highly Cooled Blunt Flat Plate at $M = 12$. NASA TN D-4320, March 1968.**

A study has been made of real-gas wedge-induced laminar separation. A series of experimental tests on a blunt flat plate with a trailing-edge flap was made at a free-stream Mach number of 12 and a free-stream Reynolds number of $10^4/\text{ft}$ ($3.3 \times 10^4/\text{m}$). The tests were conducted for stagnation enthalpies ranging from 1465 Btu/lbm (3.41 MJ/kg) to 2030 Btu/lbm (4.73 MJ/kg). A calculation technique is developed to predict the chordwise extent of wedge-induced laminar separation and agrees reasonable with the present real-gas experimental data. The computed effect of free-stream Reynolds number and total enthalpy on the extent of separation is demonstrated. Sample calculations are shown for equilibrium and frozen flow for a range of total enthalpies.

- 139. Fox, Charles H., Jr.; and Spencer, Bernard Jr.: Hypersonic Aerodynamic Characteristics of Low-Wave Drag Elliptical-Body Tail Combinations at Affected by Changes in Stabilizer Configuration. NASA TM X-1620, 1968.**

- 140. Crawford, D. D.: Pressure and Heat Transfer on Blunt Curved Plates with Concave and Convex Surfaces in Hypersonic Flow. NASA TN D-4367, March 1968.**

Pressure and heat transfer have been measured in the presence of both favorable and adverse pressure gradients at Mach numbers of 6.8 and 9.6 and Reynolds numbers based on length from 0.5×10^6 to 3.0×10^6 . The pressure data were represented by a power law relationship, and the empirical constants from this relationship were used to calculate the heat transfer by a simplified method. This method allows similar heat transfer by a simplified method. This method allows similar compressible solutions of the laminar boundary with pressure gradient to be directly applied without the use of integral or iterative methods. The results of these calculations are compared with the experimental values and with the calculated results from a more complex laminar similarity method requiring a machine program and iterative procedures. Differences between the two theoretical methods are shown to be slight for many applications. Applications where the simple method is less accurate are defined.

141. Woods, W. C.; Johnston, P. J.; Molloy, J. K.; and Henderson, A., Jr.: Recent Studies of Factors Affecting High Lift-Drag Ratios at Mach 19 in Helium. *Journal Spacecraft and Rockets*, AIAA, vol. 5, no. 1, 1968. (Also available as AIAA Paper 67-138).

Factors affecting performance of three vehicles designed for $(L/D)_{\max}$ of three and two at $M = 19$, $R = 3.2 \times 10^6$ have been examined in the Langley 22-inch helium tunnel. Two of the vehicles were initially flat-top and flat-bottom orientations of the same elementary shape. Prior to satisfying trim and stability requirements the $(L/D)_{\max}$ of the flat-bottom and flat-top shapes were 3.9 and 3.1, respectively. After satisfying the stability requirements both configurations achieved a trimmed $(L/D)_{\max}$ of about 3. The third configuration was designed to have the volume and volume distribution necessary for the convenient placement of men, stores, and equipment, and be capable of a trimmed $(L/D)_{\max}$ of 2. In this instance the geometry and center-of-gravity limits were determined by practical considerations of packaging. The result of lateral stability studies as well as other trade off considerations yielded a compromise wing-tip-fin geometry which satisfied the trim, stability, and performance requirements.

142. Henderson, A., Jr.; and Maddalon, D. V.: Hypersonic Transition Studies on a Slender Cone at Small Angles of Attack. *AIAA Journal*, vol 6, 1968, pp. 176-177.
143. Henderson, A., Jr.; and Maddalon, D. V.: Boundary-Layer Transition on Sharp Cones at Hypersonic Mach Numbers. *AIAA Journal*, vol. 6, 1968, pp. 424-431.

An experimental investigation to locate the beginning of transition from laminar to turbulent boundary-layer flow has been conducted at zero angle of attack on sharp, smooth cones having semiapex angles of 2.87° , 5° , and 10° in the contoured nozzle of the Langley 22-inch helium tunnel at a freestream Mach number of about 20. Local Mach number at the boundary-layer edge was thus varied from 7.4 to 16.6. The data indicate that local transition Reynolds number increases rapidly with local Mach number. Techniques used to detect onset of transition included surface pitot tube, drag force, boundary-layer pitot-pressure surveys, schlieren photographs, and hot-film anemometer measurements. Skin-friction coefficient, displacement thickness, momentum thickness, and velocity ratio profiles were determined for laminar, transitional, and turbulent hypersonic boundary layers. A hot-film anemometer survey of the model boundary layers showed disturbances originating within the boundary layer at much lower Reynolds numbers than the Reynolds number for which transition is felt at the model surface. The maximum disturbance level occurred at a location corresponding to about 0.84 (boundary-layer thickness) with the disturbance speed being subsonic relative to the local edge velocity. In addition, source flow effects on transition Reynolds number were examined at a local Mach number of 15.8.

- 144. Cary, Aubrey M., Jr.; and Morrisette, E. Leon: Effect of Two-Dimensional Multiple Sine-Wave Protrusions on the Pressure and Heat Transfer Distributions for a Flat Plate at Mach 6. NASA TN D-4437, March 1968.**

Surface pressure and heat transfer were measured and oil-flow patterns were observed on two-dimensional, shallow, multiple sine-wave protrusions embedded in a flat surface. The maximum laminar heating on multiple waves was found to correlate empirically with results from previous investigations. The maximum turbulent heating for a series of waves decreased rapidly from wave to wave. Tests with single waves and with the first wave of the multiple-wave model indicated that the maximum turbulent heating on single waves increased almost linearly with decreasing width-height ratio of the wave. The method used to predict the maximum turbulent heating gave fair results when there was no boundary separation prior to the wave.

- 145. Harvey, William D.; and Bushnell, Dennis M.: Velocity Fluctuation Intensities in a Hypersonic Turbulent Boundary Layer. AIAA Journal vol. 7, no. 4 p.p. 760-761, April 1968.**

The purpose of this Note is to present an estimate of the intensity of the velocity fluctuations, as based directly on the fluctuation data of Wallace and to evaluate a mixing length approach to compute indirectly the magnitude and trends of density and velocity fluctuations from measurements of mean flow quantities.

- 146. Bertram, Mitchel H.; Cary, Aubrey M., Jr.; and Whitehead, Allen H., Jr.: Experiments With Hypersonic Boundary Layers and Flow Fields, AGARD CP No. 30, May 1968, pp. 1-1-1-21.**

Recent hypersonic turbulent-boundary-layer experiments and proposed prediction methods pertinent to the problems of the effect of wall temperature on skin friction and heat transfer, the transformation of the compressible boundary layer to the constant-density type, and the heat transfer to delta wings are considered. The level of the turbulent heat-transfer coefficient is found to be little affected by significant changes in wall-temperature level. Coles' transformation as modified by Baronti and Libby has been examined by utilizing turbulent-boundary-layer profiles covering a wide range of Mach number and wall-temperature ratio. Some success is found for the transformation up to the lower end of the hypersonic range and down to moderately low wall-temperature ratios. For delta wings at low angle of attack, in cases where the flow near the surfaces is essentially streamwise, strip application of successful flatplate methods gives good predictions of the turbulent heat transfer if the pressures are known. On the lee side of delta wings where vortices are indicated, predictions by strip theory are surprisingly good in general; but predictions can be poor near the center line where the heat transfer is high. Ability to predict the heat transfer to delta wings appears to be contingent upon the ability to predict the flow field.

147. **Cary, A. M., Jr.: Turbulent Boundary Layer Heat Transfer and Transition Measurements for Cold Wall Conditions at Mach 6. AIAA Journal, May 1968.**

The purpose of the present Note is to provide additional information on turbulent boundary-layer heat transfer and transition over a wide range of wall-to-recovery temperature ratios. Tests were made in the Langley 20-inch Mach 6 wind tunnel using a flat plate that was cooled by circulating liquid nitrogen through internal coolant passages.

148. **Fox, Charles H., Jr.; and Spencer, Bernard, Jr.: Hypersonic Aerodynamic Characteristics of Low-Wave-Drag Elliptical-Body-Tail Combinations as Affected by Changes in Stabilizer Configuration. NASA TM X-1620, August 1968.**

An investigation has been made in the Langley 15-inch hypersonic flow apparatus at a Mach number of 10.03 in air to determine systematically the effects of outboard stabilizer and vertical- and vee-tail configurations on the longitudinal- and lateral-directional stability characteristics and on the resultant aerodynamic performance of a low-wave-drag elliptical body. The body had a longitudinal area distribution conforming to the theoretical shape required to minimize the zero-lift hypersonic pressure drag under the constraints of given length and volume. The body cross section was elliptical with a major-to-minor axis ratio of 2 (major axis horizontal). Bodies were tested with equivalent fineness ratios of 6.14 and 9.83. Base-mounted outboard stabilizers were tested at various dihedral angles alone and in combination with either a single center-line vertical tail or with a vee tail. The angle of attack was varied from approximately -4° to 21° at 0° and -5° of sideslip. This investigation represents the initial portion of a study to determine methods of providing stability from hypersonic through low subsonic speeds for vehicles with high hypersonic lift-drag ratios.

149. **Morrisette, E. Leon; Stone, David R.; and Whitehead, Allen H., Jr.: Boundary Layer Tripping With Emphasis on Hypersonic Flows. Presented at Symposium on Viscous Drag Reduction, Dallas, Texas, September 24-25, 1968.**

Experiments on the effect of trip geometry, size, and location on the position of transition at local Mach numbers up to 8.5 are presented. The pressure drag of the trip is investigated at local Mach numbers of 4.7 and 5.5. Based on test results, a flow model was constructed which includes trip-produced multiple vortex filaments similar to those found at supersonic speeds that are assumed to be responsible for introducing the disturbances that lead to transition.

150. **Miller, Charles G., III: Experimental Base Pressures on 9° Spherically Blunted Cones at Mach Numbers From 10.5 to 20. NASA TN D-4800, October 1968.**

Base pressure measurements were obtained in the Langley hotshot tunnel at an angle of attack of 0° for a series of cones having semiapex angles of 9° and bluntness ratios of 0, 0.3, 0.55, and 0.8. Nominal free-stream Mach numbers were 10.6, 13.8, 15.6, and 19.6 and free-stream Reynolds numbers, based on model surface length, were approximately 0.02×10^6 to 2×10^6 in nitrogen. These conditions were assumed to result in laminar boundary-layer flow. Sting surface pressure distributions and sting tuft flow patterns were obtained 0 to 3.75 model base diameters downstream of the model base.

151. **Edwards, Clyde L. W.; and Anders, John B.: Low Density Leading Edge Bluntness and Ablation Effects on Laminar Ramp-Induced Boundary Layer Separation at Moderate Enthalpies. NASA TN D-4829, October 1968.**

A study of real-gas wedge-induced laminar-boundary-layer separation has been made. The investigation was conducted in low-density air on a highly cooled flat-plate model with interchangeable leading edges and various trailing-edge flap angles. All tests were conducted in the Langley 1-foot hypersonic arc tunnel at a nominal free-stream Mach number of 12, free-stream unit Reynolds numbers from 1.1×10^4 to 2.7×10^4 per foot (3.6×10^4 to 8.9×10^4 per meter), and dimensionless stagnation enthalpies from 39.0 to 72.4. The extent of separation was found to increase with increasing leading-edge bluntness at these test conditions. Significant low-density effects are shown to delay the onset of separation in comparison with what would be expected from predictions by a strong-interaction theory. The direct effect of mass addition on the extent of separation through ablative leading edges was found to be negligible; however, an indirect effect on the extent of separation due to leading-edge regression was found to be significant.

152. **Whitehead, A. H., Jr.; and Keyes, J. W.: Flow Phenomena and Separation Over Highly Swept Delta Wings With Trailing Edge Flaps at Mach 6. AIAA Journal, vol. 6, no. 12, December 1968, pp. 2380-2387.**

The hypersonic flow field over highly swept delta wings with various types of separation is investigated. Heat-transfer rates, pressure distributions, and several flow-visualization techniques were used to experimentally examine the flow over a large angle-of-attack range. It was found that where the boundary-layer type differs across the span prior to separation, complex and unusual flow phenomena develop. When the boundary layer is turbulent over the span of the wing prior to separation, the surface heating and pressures can be estimated by two-dimensional calculations over the wing and flap. At small angles of attack, separation can occur on the lee surface either inboard of, or at the leading edge depending on vehicle geometry and test conditions. By means of several flow-visualization techniques, the separate flow is found to form coiled vortices in a manner similar to that found at subsonic speeds but with several significant differences.

153. Love, E. S.; Woods, W. C.; Rainey, R. W.; and Ashby, G. C., Jr.: **Some Topics in Hypersonic Body Shaping.** Presented at AIAA 7th Aerospace Sciences Meeting, January 1969.

Some topics related to hypersonic body shaping for minimum drag and improved performance are presented. Implications of the most frequently assumed pressure laws are reviewed from the view of practical flight regions, and solutions expressed in exponential residuals are presented for inviscid minimum drag power-law bodies (simple and complex) for a wide range of fineness ratio. Constraints considered are length-and-diameter, and length-and-volume. Comparisons are made with experiment. Experimental results from Mach 6 to 20 are presented for a series of trapezoidal bodies; the cross-section of the body giving best lift-drag ratio differs from that indicated in earlier work. A simple "sin²-deficiency method" is presented for predicting pressures on blunt shapes; the method appears to yield results comparable to more elaborate methods requiring machine computation.

154. Henderson, A., Jr.: **Hypersonic Viscous Flows.** Modern Developments in Gas Dynamics, Plenum Press, pp. 83-129, NASA TN D-4961, 1969.
155. Harvey, William D.; Bushnell, Dennis M.; and Beckwith, Ivan E.: **On the Fluctuating Properties of Turbulent Boundary Layers for Mach Numbers Up to 9.0.** NASA TN D-5496, 1969.
156. Harvey, W. D.; and Bushnell, D. M.: **Estimated Levels of Velocity Fluctuating Intensities in Hypersonic Turbulent Boundary Layer.** AIAA Journal, vol. 7, no. 4, April 1969, pp. 760-762.
157. Wagner, R. D.; and et al.: **Influence of Measured Free-Stream Disturbance on Hypersonic Boundary Layer Transition.** AIAA Paper No. 69-704, June 1969.

Results of a study of transition in hypersonic flow over a wide range of test conditions in the Langley M = 20 helium tunnel are presented. Direct measurements of the facility free-stream disturbances have been made with a constant current anemometer which has a frequency response capability of 500 kilohertz. In unheated flow, fluctuation mode diagrams identify the disturbances as sound waves produced by moving sources of sound (the turbulent nozzle wall boundary layer). The test section disturbance levels can be quite high with a maximum of about 3.5 percent rms mass flow fluctuation (which corresponds to about 6 percent rms static pressure fluctuation) when transition of the nozzle wall boundary layer occurs in the portion of the nozzle which and influence the test section disturbance level via direct acoustic radiation. For the same range of test conditions, transition measurements have been made on a 10° half-angle wedge. (Extensive amounts of laminar, transitional, and turbulent flow data are presented.)

158. **Goldberg, Theodore J.; Hefner, Jerry N.; and Stone, David R.: Hypersonic Aerodynamic Characteristics of Two Delta-Wing X-15 Airplane Configurations. NASA TN D-5498, October, 1969.**

An investigation of the hypersonic aerodynamic characteristics of two delta-wing X-15 research configurations was conducted at a Mach number of 6 and a Reynolds number of 8.27×10^6 per foot. Limited tests of one model were also made at a Mach number of 8. Results are presented to show the aerodynamic effects of wing geometry and longitudinal position, wing fins, nose cant, strakes, speed brakes, and a suspended test ramjet. The type of boundary layer ahead of the elevons was determined from oil-flow studies of the separation boundaries with and without boundary-layer trips. Experimental aerodynamic characteristics are compared with analytical estimates.

159. **Fox, Charles H.; and Spencer, Bernard Jr.: Transonic Aerodynamic Characteristics of Hypersonic Low-Wave Drag Elliptical-Body-Tail Combinations as Affected by Changes in Stabilizer Configuration. NASA TM X-1789, 1969.**

160. **Miller, Charles G.; and Smith, Fred M.: Measured Base Pressure on a Blunted Cone at Mach Numbers 10 to 76. NASA TM X-62828, March 1970.**

161. **Fischer, Michael C.: An Experimental Investigation of Boundary-Layer Transition On A 10° Half-Angle Cone At Mach 6.9. NASA TN D-5766, April, 1970.**

An experimental investigation was made of the effect of mass addition, local unit Reynolds number, nose bluntness, and angle of attack on the transition Reynolds number for a 10° half-angle cone with a ratio of wall to total temperature of about one-half. The tests were conducted at a Mach number of 6.9 and a free-stream unit Reynolds number range of 1.68×10^6 to 6.26×10^6 per foot. The results showed a significant reduction in transition Reynolds number owing to mass addition by using a low-temperature ablator. There was a significant influence of local unit Reynolds number on transition. The present data did not agree with a correlation which attempts to predict the Reynolds number for the end of transition. Correlations of transition Reynolds number with hypersonic Mach number were shown to be highly dependent on tunnel size (noise level). Transition moved forward on the leeward side of the sharp cone at angle of attack and rearward on the windward side. Transition occurred only on the leeward side of the blunted-cone configurations at angle of attack and was displaced rearward of equivalent sharp-cone transition location. At large angle of attack there was a diminishing effect of bluntness on displacing transition rearward. Longitudinal grooves observed on the surface of the ablated models were believed to be formed by an array of streamwise Gortler-type vortices.

162. **Bertram, Mitchel H.; and Henderson, Arthur, Jr.: Some Recent Research With Viscous Interacting Flow in Hypersonic Streams. NASA TM X-66914, 1970.**

- 163. Cary, Aubrey M., Jr.: Turbulent Boundary Layer Heat Transfer and Transition Measurements With Extreme Surface Cooling at Mach 6. NASA TN D-5863, June 1970.**

The effect of wall cooling on turbulent boundary-layer heat transfer has been a topic of dispute since X-15 flight results indicated that the turbulent heat-transfer coefficient was virtually independent of the wall-to-recovery temperature ratio. Recent results obtained in ground facilities have tended to substantiate the flight measurements. The purpose of the present Note is to provide additional information on turbulent boundary-layer heat transfer and transition over a wide range of wall-to-recovery temperature ratios.

- 164. Small, William J.; Kirkham, Frank S.; and Fetterman, David E.: Aerodynamic Characteristics of Hypersonic Transport Configuration Vehicle Design at Mach 6.86. NASA TN D-5885, June 1970.**

An investigation of model representative of a hypersonic transport has been conducted at a Mach number of 6.86 over a range of Reynolds numbers, based on body length, of 1×10^6 to 6×10^6 . The configuration was a low-wing, distinct wing-body arrangement with a body-mounted vertical tail and an underwing propulsion system. The complete vehicle and the contribution of its components are analyzed in order to evaluate the performance of this class of vehicle. Present methods of predicting aerodynamic performance were also evaluated.

- 165. Townsend, James C.: Aerodynamic Interference Effects on Half-Cone Bodies with Thin Wings at Mach 10.03. NASA TN D-5898, July 1970.**

- 166. Whitehead, A. H., Jr.: Vortices in Separated Flows. AIAA Journal, vol. 8, June 1970, pp.1173-1174.**

- 167. Watson, R. D.; and Weinstein, L. M.: A Study of Hypersonic Corner Flow Interactions. AIAA Paper 70-227, 1970.**

Extensive measurements at Mach 20 in helium of the flow characteristics over sharp leading-edge 90° corner flow models are presented. The measurements include heat transfer and surface pressure distributions, pitot surveys in the base plane of the models, oil-flow photographs, and electron beam low visualization photographs. In addition, heat-transfer distributions are presented for sharp flat-plates intersecting at angles of 60° , 120° , and 270° at Mach 8 in air.

168. Keyes, J. Wayne and Watson, Ralph D.: **Laminar Heating in Interior Corners at Mach 19.** *Journal of Spacecraft and Rockets*, vol. 7, no. 8, August 1970, pp. 1012-1013.

Hypersonic vehicle designs require a knowledge of the interacting flow field in the vicinity of interior corners as, for example, at wing-fuselage junctures and in two-dimensional inlets. Experimental results at Mach 3 and Mach 8 have shown a complicated corner flow structure exists with surface heat transfer and pressure distributions significantly different from local wedge or plate values. Recent studies made at Mach 20 considered symmetrical corners; however, in practical cases one generally encounters asymmetrical corners. The present study examines heat transfer and surface oil flow on symmetrical and asymmetrical corner configurations at Mach 19 in the Langley 22-inch helium tunnel at a freestream Reynolds number of approximately 1.5×10^6 .

169. Beckwith, Ivan E.; Harvey, William D.; and Clark, F. L.: **Comparisons of Turbulent Boundary Layer Measurements at Mach Number 19.5 With Theory and an Assessment of Probe Errors.** NASA TN D-6192, 1971.

170. Dillon, James L.: **Analysis of Surface Pressure on a 5° Cone in Free-Flight Near Mach 20 (Reentry F).** NASA TM X-2210, February 1971.

Pressure measurements were made at 13 locations on a 396-cm-long (156-inch) 5° half-angle conical spacecraft during reentry at a free-stream Mach number near 20. The cone surface was beryllium except for the graphite nose which had an initial tip radius of 0.25 cm (0.10 inch). The angle of attack was less than 1° during the entry from 30.48 km to 15.24 km (100,000 to 500,000 feet). Comparison of theory with data measured along the spacecraft indicated that sharp cone pressure existed over the rearward two-thirds of the cone. In most cases, theoretical pressures in this region, computed with the tangent cone concept, were within the accuracy band of the measured data. An attempt was made to establish a nose radius history from data that were measured in the region influenced by nose bluntness. The trends and magnitude of the circumferential pressure data are represented reasonably well by the theory of High and Blick.

171. Carter, Howard S.; Raper, James L.; Hinson, William F.; and Morris, W. Douglas: **Spacecraft Description and Basic Measurements From a Flight Experiment to Determine Turbulent Heating and Boundary Layer Transition at Local Mach Numbers Near 15 (Reentry F).** NASA TM X-2308, 1971.

172. Bushnell, Dennis M.; and Morris, Dana J.: **Eddy Viscosity and Mixing Length Distributions in Hypersonic Turbulent Boundary Layers.** NASA TM X-2310, 1971.

173. Hefner, J. N.: **Boundary-Layer Transition for Space-Shuttle-Type Configurations at Angles of Attack.** NASA TM X-2254, March 1971.

174. Bushnell, D. M.; and Morris, D. J.: **Eddy Viscosity Distributions in a Mach 20 Turbulent Boundary Layer.** Technical Note, *AIAA Journal*, vol. 9, no. 4, April 1971, pp. 764-766.

This Note presents shear stress and eddy viscosity distributions obtained from this high Mach number profile data and indicates the most consistent of the available models for turbulent shear.

175. Blackstock, T.; Gross, D. L.; and Rister, W.: **Static Aerodynamic Stability and Control Characteristics of a GD/Convair Delta Wing Booster at Nominal Mach Number = 10.** SADSAC DMS-DR-1093, May 1971.
176. Wright, Robert L.; and Zoby, Ernest V.: **Flight Measurements of Boundary-Layer Transition on 5° Half-Angle Cone at Free Stream Mach Numbers of 20 (Reentry F).** NASA TM X-2253, May 1971.
177. Watson, R. D.; and Weinstein, L. M.: **A Study of Hypersonic Corner-Flow Interactions.** AIAA Journal, vol. 9, no. 7, July 1971, pp. 1280-1286.

Characteristics of the hypersonic flow fields over three sharp leading-edge internal corner models have been measured at Mach 20 in helium. Wedges of equal angles (0°-0°, 5°-5°, and 10°-10°) intersecting at 90° formed the models. The measurements include heat-transfer and surface pressure distributions, Pitot surveys in the base plane of the models, oil-flow photographs, and electron beam flow visualization photographs. The data indicate that the broad features of external and internal shock structure observed at supersonic Mach numbers also occur at Mach 20; however, the presence of large vortices and thick boundary layers distort the flow in the immediate vicinity of the corner. Observed peaks in surface heat transfer correlate with a strong vortex near the corner and a disturbance propagated from the inviscid flow into the boundary layer. The flow field appears to be basically conical in nature except for large values of X the hypersonic viscous interaction parameter. Corner heating rates, relative to undisturbed wedge or flat-plate heating rates, increase significantly with increasing freestream Mach number.

178. Goldberg, T. J.; and Hefner, J. N.: **Starting Phenomena for Hypersonic Inlets with Thick Turbulent Boundary Layers at Mach 6.** NASA TN D-6280, August 1971.

The effects of cowl length, cowl bluntness, cowl angle of attack, boundary-layer thickness, free-stream Reynolds number, and wall temperature on the starting phenomena of two-dimensional hypersonic inlets with turbulent intake boundary layers were experimentally investigated at a free-stream Mach number of 6. The inlet total-pressure recovery (including both shock and viscous losses) governs starting in contrast to only the normal-shock pressure recovery usually considered. The total-pressure recovery required for starting is predicted reasonably well by a one-dimensional analysis.

179. Beckwith, Ivan E.; and Bertram, Mitchel: **NASA Langley Studies on Stability and Transition in High-Speed Shear Layers.** Presented at the 1971 Boundary Layer Transition Specialists Workshop, San Bernadino, California, November 3-5, 1971.

- 180. Wagner, R. D.: Hot Wire Measurements of Freestream and Shock Layer Disturbance. AIAA Journal, vol. 9, no. 12, Dec. 1971, pp. 2468-2470**

Wind-tunnel freestream disturbances have been speculated to be a prominent factor in hypersonic boundary-layer transition. In order to obtain a quantitative assessment of the role of the facility disturbance level in hypersonic boundary-layer transition, future transition studies must include measurements of the disturbance environment. However the problem remains to determine whether disturbance measurements in the freestream alone are adequate to describe the model boundary-layer input disturbances, or does the model shock wave change the freestream disturbances before they reach the model.

- 181. Fischer, M. C.; and Wagner, R. D.: Effects of Wind Tunnel Disturbance on Hypersonic Boundary Layer Transition. AIAA Paper 72-181, January 1972.**

An experimental investigation of the dependence on facility disturbance of transition on sharp cones has been made. The sound radiated from the turbulent boundary layer on the walls of several facilities was measured with pressure transducers mounted flush with the cone surface. A constant current hot-wire anemometer measured free-stream and cone shock layer disturbances in two hypersonic helium tunnels. Comparison of hot-wire results with surface pressure measurements in one of the helium tunnels indicated the latter data provide an accurate indication of the facility disturbance levels. Both studies show that transition Reynolds numbers correlate in terms of facility rms sound disturbance levels provided the laminar boundary layers on the models are similar.

- 182. Dillon, James L.; and Carter, H. S.: Analysis of Base Pressure on Base Heating on a 5° Half-Angle Cone in Free Flight Near Mach 20 (Reentry F). NASA TM X-2468, January 1972.**

Heating and pressure measurements were made on the base of a 396-centimeter-long 5°-half-angle conical spacecraft during reentry at a free-stream Mach number near 20 (Reentry F). The cone surface was beryllium except for the graphite nose which had an initial tip radius of 0.25 centimeter. Angle of attack was less than 1° during the entry from 30.48 kilometers to 15.24 kilometers. The predicted values of pressure from an extrapolation of Cassanto's turbulent correlation were lower than the measured data except at the lower altitudes. The trend of the laminar heating data and the turbulent data at the highest Reynolds numbers was represented reasonably well by two semiempirical theories. A laminar correlation by King underpredicted the laminar heating data.

- 183. Ashby, George C., Jr.: Comparison of Hinge Moments for a Simple Delta Wing and a Delta Wing Orbiter Concept at Mach 6. NASA TN D-6657, February 1972.**

Eleven hinge moments were determined from measured surface pressures on a typical delta-wing shuttle orbiter model at selective deflection angles for comparison with the extensive experimental and analytical hinge-moment previously reported for a simple 75° delta wing with a trailing-edge control. The angles of attack were from 0° to 55° at elevon-deflection angles of 145.5° , 0° , and 20° . The results show that the elevon hinge moments on the shuttle orbiter are essentially the same as those measured earlier for the more basic model. Also included is an appendix describing a cubic spline function technique used to determine the hinge moments from elevon surface-pressure measurements.

- 184. Beckwith, Ivan E.; and Bertram, Mitchel H.: A Survey of NASA Langley Studies on High Speed Transition and the Quiet Tunnel. NASA TM X-2566, July 1972.**

The present studies include a quantitative experimental and theoretical assessment of the role of wind-tunnel disturbances in the boundary-layer transition process at hypersonic speeds. The various approaches and recent results for the development of a low-noise-level tunnel are presented. A statistical parametric study of transition data with a large computer is shown for ones in free flight, ballistic ranges, and wind tunnels at essentially zero angle of attack. New transition results on slender cones at small angle of attack are also given, as are studies of transition at high angle of attack, which are compared with various correlation attempts. Included are results which indicate that hypersonic transition in the outer part of the boundary layer precedes the manifestation of transition at the wall ("precursor" transition).

- 185. Birch, S. F.; and Keyes, J. W.: Transition in Compressible Free Shear Layers. Journal of Spacecraft and Rockets, AIAA Paper, vol. 9, August 1972, pp. 623-624.**

The results presented here are based on two separate studies using the Langley 20-inch (Mach 6) and 11-inch (Mach 6.9) Hypersonic Tunnels. In each case, a variable angle wedge generates a planar shock wave which interacts with the bow wave of a bluff body. The interaction geometry obtained in the two facilities differed in that the bluff body used in the 11-inch tunnel was two-dimensional, 7.62 cm (3-in.) long, and 6.35 cm (2 1/2-in.) wide, whereas that used in the 20-inch tunnel was a hemisphere/cylinder, 5.08 cm (2-in.) in diam.

- 186. Arrington, James P.; and Ashby, George C., Jr.: Effect of Configuration Modifications on the Hypersonic Aerodynamic Characteristics of a Blended Delta Wing Body Entry Vehicle. NASA TM X-2611, September 1972.**

This study includes the longitudinal, lateral, and directional aerodynamic characteristics of a delta-wing configuration obtained experimentally at Mach 20 in helium with Reynolds numbers, based on model length, of 1.5×10^6 and 2.9×10^6 and at a Mach number of 6 in air with a Reynolds number, based on model length, of 4.8×10^6 . The angles of attack varied from 0° to 55° for two sideslip angles. The effects of the addition of dorsal fins, the removal of wing tip fins, an increase in elevon span, and changes in elevon hinge-line sweep angle are discussed. The unmodified vehicle had a maximum lift-drag ratio of 2.1 at Mach 19 and of 2.4 at Mach 6 with about the same lateral and directional stability level at both Mach numbers.

- 187. Stainback, P. C.; and Wagner, R. D.: A Comparison of Disturbance Levels Measured in Hypersonic Tunnels Using a Hot Wire Anemometer and a Pitot Pressure Probe. AIAA Paper 72-1003, September 1972.**

Disturbance levels were measured in the test section of a Mach 5 blowdown jet using a constant-current, hot-wire anemometer and a pressure transducer. The pressure transducer was mounted flush with the end of a tube and oriented to measure the fluctuating pitot pressures. The disturbance levels, measured by the two instruments and normalized by local mean values, agreed within about 30 percent where the pitot data were higher than the hot-wire data. The rms disturbance levels measured with the hot-wire anemometer and converted to pitot pressures using a quasi-steady flow analysis, were about two-thirds the levels measured with the pitot probe. The variation of the normalized rms disturbance levels with stagnation pressure indicated that transition occurred in the boundary layer on the nozzle wall and influenced the outputs of the instruments located at the exit of the nozzle when the total pressure was about 35 N/cm^2 . Below this pressure, the disturbance levels decreased markedly. At higher pressures, the disturbances were predominantly aerodynamic noise generated by the turbulent boundary layer on the nozzle wall.

- 188. Hefner, J. N.; and Sterrett, J. R.: Secondary Jet Interaction With Emphasis on Outflow and Jet Location. Journal of Spacecraft and Rockets, vol. 9, November 1972, pp. 845-847.**

Data were obtained concerning a model which utilized a two-dimensional sonic jet. It was found that outflow and jet location significantly influence the magnitude and behavior of the secondary jet interaction forces. Continued refinements in jet interaction analysis which do not account for outflow are of limited usefulness in the design of control systems for supersonic and hypersonic vehicles. The jet total back pressure ratio increases with either jet pressure ratio or freestream Mach number.

189. **Holla, V. S.; Subash, A. N.; and Kumar, A.: Evaluation of Lift and Drag of a Family of Conically Cambered Wings at off Design Mach Numbers. Journal of the Aeronautical Society of India, vol. 24, no. 4, November 1972, pp. 390-402..**

In this paper a method for evaluating the lift and drag at off design Mach numbers of a family of conically cambered wing with subsonic leading edges and supersonic trailing edge is given. The method is valid only for small changes in the flight Mach number. A simple rule of thumb method is also given to obtain the expression for lift dependent drag at off design Mach number.

190. **Wagner, Richard D.: Measured and Calculated Mean Flow Properties of a Two-Dimensional Hypersonic Turbulent Wake. NASA TN D-6927, November 1972.**

The hypersonic turbulent wake produced by a wedge was studied experimentally and its properties were compared with predictions obtained from a numerical computation procedure. In the computation procedure several models for the eddy-viscosity formulation of the turbulent transport were examined. Conventional-defect models and a modified mixing-length model were found to yield good predictions of the experimental data. The classical mixing-length model gave unrealistic results. The experimental data displayed similarity when velocity and temperature defects were scaled by the maximum defects and the transverse coordinate was scaled by the velocity-defect half-width.

191. **Rao, D. M.; and Whitehead, A. H., Jr.: Lee-Side Vortices on Delta Wings at Hypersonic Speeds, AIAA Journal, vol. 10, November 1972, pp. 1458-1465.**

A fluid-dynamic investigation was carried out to determine the cause of intense heating observed on the lee meridian of hypersonic delta wings and also to derive means for its suppression. Several experimental techniques were combined with analysis of extensive heat-transfer measurements at a freestream Mach number of six in a range of Reynolds number to acquire a general description of the lee-flow structure. With attached leading-edge flow on the delta wings, the dominant feature is a pair of embedded vortices on the lee meridian whose interaction with the boundary-layer is responsible for the observed local heating. On the basis of flow visualization results and heat-transfer correlations, a qualitative vortex flow model is proposed which differs essentially from the conventional inboard separation vortex model.

192. **Hefner, Jerry N.; and Whitehead, Allen H., Jr.: Surface-Flow, Pressure, and Heat-Transfer Studies on Two Conical Delta Wings at a Mach Number of 6. NASA TM X-2668, December 1972.**

An experimental investigation of the surface flow, pressures, and heat transfer on two conical delta wings having attached leading-edge shocks has been conducted at a Mach number of 6. The angle of attack was varied between 0° and 12° . The pressure data were compared with predictions obtained by the method-of-lines technique, and the heating data were compared with the heating level predicted by the Spalding-Chi method.

193. **Waston, R. D.; Harris, J. E.; and Anders, J. B., Jr.: Measurements in a Transitional/Turbulent Mach 10 Boundary Layer at High Reynolds Number. AIAA Paper 73-166, January 1973.**
194. **Wagner, R. D.: Mean Flow and Turbulence Measurements in a Mach 5 Shear Layer. Part II - Hot Wire Measurements of the Mean Flow and Turbulence Intensities. Fluid Mechanics of Mixing, ed. by E. M. Uram and V. W. Goldschmidt, ASME, pp. 79-82, June 1973.**

There exists a continuing interest in free turbulent mixing in supersonic and hypersonic flows due to the large number of possible technology applications such as propulsion, shock interference heating, wakes, noise reduction efforts, and slot injection film cooling. While many low-speed studies are available, only in the past 8 years have some free turbulent shear layer data become available for Mach numbers greater than 2.0. The data presented herein result from a study of turbulent mixing in the near field of a Mach 5 jet and extend both mean and fluctuating measurements in the free shear layer of the 10.6 cm. diameter jet. The jet was enclosed in a 61- by 61-cm vacuum chamber and exhausted into a diffuser 44 cm downstream of the jet exit. An auxiliary air supply to the chamber was used to equalize the chamber and nozzle static pressures. The nozzle wall boundary layer was turbulent at the jet exit for the higher Reynolds number data reported herein.

195. **Arrington, James P.; and Stone, David R.: Aerodynamic and Flow Visualization Studies of Two Delta-Wing Entry Vehicles at a Mach Number of 20.3. NASA TN D-7282, August 1973.**
196. **Wagner, Richard D.: Mean Flow Turbulence Measurements in a Mach 5 Free Shear Layer. NASA TN D-7366, December 1973.**

A study of the time-averaged mean flow and the turbulence in a Mach 5 free turbulent shear layer has been performed. When the experimental data were reduced with the assumption of constant static pressure through the shear layer, the mean-velocity profile in similarity coordinates was in good agreement with the low-speed velocity profile. The intensities of the velocity fluctuations occurring in the same regions of the supersonic and low-speed shear layers. A large density fluctuation was observed in the outer part of the shear layer near the boundary of the shear layer and the potential core.

- 197. Townsend, James C.: Hypersonic Aerodynamic Characteristics of a Family of Power-Law, Wing-Body Configurations. NASA TN D-7427, December 1973.**

The configurations analyzed are half-axisymmetric, power-law bodies surmounted by thin, flatwings. The wing planform matches the body shock-wave shape. Analytic solutions of the hypersonic small disturbance equations form a basis for calculating the longitudinal aerodynamic characteristics. Boundary-layer displacement effects on the body and the wing upper surface are approximated. Skin friction is estimated by using compressible, laminar boundary-layer solutions. Good agreement was obtained with available experimental data for which the basic theoretical assumptions were satisfied. The method is used to estimate the effects of power-law, fineness ratio, and Mach number variations at full-scale conditions.

- 198. Stainback, P. Calvin; and Wagner, Richard D.: Experimental Studies of Hypersonic Boundary Layer Transition and Effects of Wind Tunnel Disturbances. NASA TN D-7453, 1974.**

Boundary-layer transition data on cones and free-stream disturbance levels were measured in the Ames 3.5-foot hypersonic wind tunnel and the Langley Mach 8 variable density hypersonic tunnel. Transition data were obtained by using different conical models and techniques for detecting the location of transition. The disturbance levels were measured by using hot-wire anemometry transducers. The transition Reynolds numbers obtained from the tests at the Ames Research Center correlated well with other transition data obtained in similar facilities at the Langley Research Center when the fluctuating pressures measured at the surface of conical models were used as correlating parameter.

- 199. Owen, F. K.; Horstman, C. C.; Stainback, P. C.; and Wagner, R. D.: Comparison of Transition and Free Stream Disturbance Measurements Obtained in Two Wind Tunnel Facilities. AIAA Paper 74-131, January 1974.**

Boundary-layer transition measurements have been made on two 5-degree half-angle cones at Mach 7 in the Ames 3.5-foot and the Langley variable density wind tunnels. Although there were differences between the measured free-stream disturbance scales and pressure fluctuation levels, the choice of consistent locations within the transition region, using either thin-film fluctuation or surface heat-transfer data results in excellent agreement between changes in free-stream pressure fluctuation location in the two facilities.

- 200. Watson, Ralph D.: Experimental Study of Sharp and Blunt Nose Streamwise Corners at Mach 20 — Hypersonic Shock-Boundary Layer Interaction Parameters. NASA-TN D-7398, April, 1974.**

An analytical and experimental investigation has been made to provide a space shuttle orbiter wing design that met the guideline requirements of landing performance, stability, and hypersonic trim for a specified center-of-gravity envelope. The analytical study was facilitated by the use of the Optimal Design Integration system (DDIN) and the experimental part of the investigation was conducted in the Langley low-turbulence pressure tunnel and the Langley continuous-flow hypersonic tunnel.

201. Hefner, Jerry N. and Cary, Aubrey M., Jr.: Swept-Slot Film-Cooling Effectiveness in Hypersonic Turbulent Flow. AIAA Journal of Spacecraft and Rockets, vol. 11, no. 5, pp. 351-352. Journal of Spacecraft and Rockets, vol. 11, May 1974, pp. 361-362.

Film cooling provides a means of reducing the operational surface temperature of a high-speed vehicle below the radiation equilibrium temperature. Experimental studies of tangential slot injection at Mach 6 have shown that the film-cooling effectiveness in a two-dimensional, high-speed turbulent flow is significantly greater than indicated by extrapolations of previous low-speed results. However, practical applications of slot injection film cooling (particularly on wings) will generally require the slots to be swept relative to the inviscid streamline direction. The effect of sweeping the slot on the film-cooling effectiveness downstream of the slot has not been previously investigated. The present Note presents measurements of surface equilibrium temperature downstream of swept slots with sonic tangential air injection into a thick hypersonic turbulent boundary layer and compares these results with unswept slot results.

202. Ashby, George C., Jr.; and Harris, Julius E.: Boundary-Layer Transition and Displacement-Thickness Effects on Zero-Lift Drag of a Series of Power-Law Bodies at Mach 6. NASA TN D-7723, July 1974.

Wave and skin-friction drag have been numerically calculated for a series of power-law bodies at a Mach number of 6 and Reynolds numbers, based on body length, from 1.5×10^6 to 9.5×10^6 . Pressure distributions were compared on the nose by the inverse method and on the body by the method characteristics. The pressure distributions and the measured locations of boundary-layer transition were used in a nonsimilar-boundary-layer program to determine viscous effects. A coupled iterative approach between the boundary-layer and pressure-distribution programs was used to account for boundary-layer displacement-thickness effects. The calculated-drag coefficients compared well with previously obtained experimental data.

203. Ashby, George C, Jr.: Longitudinal Aerodynamic Performance of a Series of Power-Law and Minimum Wave-Drag Bodies at Mach 6 and Several Reynolds Numbers. NASA TM X-2713, August 1974.

Experimental data have been obtained for two series of bodies at Mach 6 and Reynolds numbers, based on model length, from 1.4×10^6 to 9.5×10^6 . One series consisted of axisymmetric power-law bodies geometrically constrained for constant length and base diameter with values of the exponent n of 0.25, 0.5, 0.6, 0.667, 0.75, and 1.0. The other series consisted of positively and negatively cambered bodies of polygonal cross section, each giving a constant longitudinal area distribution conforming to that required for minimizing zero-lift wave drag at hypersonic speeds under the geometric constraints of given length and volume.

204. **Watson, R. D.: Wall Cooling Effects on Hypersonic Transition/Turbulent Boundary Layers at High Reynolds Numbers. Presented at AIAA 8th Fluid and Plasma Dynamics Conference, Hartford, CT, June 16-18, 1975.**

The characteristics of a thick hypersonic boundary layer that is turbulent for a length of 175 cm on a 4° sharp wedge have been measured. The resulting boundary layer was free from transverse curvature effects and only mildly affected by upstream history effects caused by pressure and wall temperature gradients. Heat-transfer distributions were used to locate regions of laminar, transitional, and turbulent flow at an edge unit Reynolds number of $0.47 \times 10^6/\text{cm}$ at wall-to-total temperature ratios from about 0.3 to 1. Wall cooling had little effect on the location of the transition region. Pitot and total temperature profiles and skin-friction measurements also were obtained at several locations along the longitudinal centerline of the model. Mixing length and turbulent Prandtl number distributions were derived from the fully turbulent mean profiles.

205. **Bushnell, D. M.; Cary, A. M., Jr.; and Holley, B. B.: Mixing Length in Low Reynolds Number Compressible Turbulent Boundary Layers. AIAA Journal, Technical Note, vol. 13, no. 8, August 1975, pp. 1119-1121.**

The purpose of the present Note is to re-examine the question of low Reynolds number effects in high-speed turbulent boundary layers, and in particular, to determine whether low Reynolds number amplification of shear stress is a result of transitional flow structure.

206. **Rao, D. M.: Hypersonic Incipient Separation on Delta Wing with Trailing-Edge Flap. AIAA Journal, vol. 13, October 1975, pp. 1386-1388.**

The paper reviews the experimental data on the incipient separation characteristics of planar delta wings of 75° sharp leading edges, with full-span trailing-edge flap deflected into the windward flow. The local Reynolds number range for these investigations covered laminar, transitional, and turbulent conditions. It is shown that, while turbulent boundary layer data correlates with two dimensional results, in the laminar and transitional cases, there is a nearly parallel shift to higher flap angles for incipient separation.

207. **Kruse, Robert L.; Lovett, George H.; and Spencer, Bernard, Jr.: Reynolds Number Effects on the Aerodynamic Characteristics of Irregular Planform Wings at Mach Number of 0.30. NASA TM X-73132, 1977.**

208. **Clark, Louis E.; and Richie, Christine B.: Aerodynamic Characteristics at Mach 6 of a Hypersonic Research Airplane Concept Having a 70 Swept Delta Wing. NASA TM X-3475, May 1977.**

An experimental investigation was conducted at Mach 6 to determine the hypersonic aerodynamic characteristics of an air-launched, delta-wing research aircraft concept. Included was the effect of various components such as nose shape, wing camber, wing location, center vertical tail, wing tip fins, forward delta wing, engine nacelle, and speed brakes. Tests were conducted with a 0.021-scale model at a Reynolds number, based on model length, of 10.5×10^6 and over an angle-of-attack range from -4 to 20.

- 209. Watson, R. D.: Wall Cooling Effects on Hypersonic Transitional/Turbulent Boundary Layers. AIAA Journal, vol. 15, no. 10, October 1977, pp. 1451-1461.**

The characteristics of a thick hypersonic boundary layer that is turbulent for a length of 175 cm on a 4° sharp wedge have been measured. The resulting boundary layer was free from transverse curvature effects and only mildly affected by upstream history effects caused by pressure and wall temperature gradients. Heat-transfer distributions were used to locate regions of laminar, transitional, and turbulent flow at an edge unit Reynolds number of $0.47 \times 10^6/\text{cm}$ at wall-to-total temperature ratios from about 0.3 to 1. Wall cooling had little effect on the location of the transition region. Pitot and total temperature profiles and skin-friction measurements also were obtained at several locations along the longitudinal centerline of the model. Mixing length and turbulent Prandtl number distributions were derived from the fully turbulent mean profiles.

- 210. Watson, Ralph D.: Characteristics of Mach 10 Transitional and Turbulent Boundary Layers. NASA TP-1243, November 1978.**

Measurements of the mean-flow properties of transitional and turbulent boundary layers in helium on 4° and 5° wedges have been made for flows with edge Mach numbers from 9.5 to 11.3, ratios of wall temperature to total temperature of 0.4 to 0.95, and maximum length Reynolds numbers of 100×10^6 . The data include pitot and total-temperature surveys and measurements of heat transfer and surface shear. In addition, with the assumption of local similarity, turbulence quantities such as the mixing length were derived from the mean-flow profiles and compared with other data and theory. Low Reynolds number and precursor transition effects were significant factors at these test conditions and were included in finite-difference boundary-layer predictions.

- 211. Watson, Ralph D.: Exploratory Investigation of the Effect of Porous Surfaces on Hypersonic Turbulent Boundary Layers. NASA TM-78791, 1979.**

An exploratory investigation has been made with four different test surfaces to determine the effect of porous walls on the characteristics of a hypersonic turbulent flat-plate boundary layer. The investigation was an attempt to decrease the surface shear of the boundary layer by transmitting fluctuating pressure energy through the wall and absorbing it in a cavity beneath the wall. Pitot surveys were made at two locations schliern photographs of the boundary layer were taken, and hot-wire measurements in the flow above the boundary layer were made to determine whether the surfaces affected the turbulent boundary layer. Neither an increase nor a decrease in surface shear or boundary-layer growth was detected. Tabulations of the pitot data are included.

212. **Miller, Charles G. III; and Gnoffo, Peter A.: Pressure Distributions and Shock Shapes for 12.84°/7° On-Axis and Bent-Nose Bionics in Air at Mach 6. NASA TM X-83222, December 1981.**

Pressure distributions and shock shapes on a spherically blunted, 12.84 deg/7 deg on axis biconic and a spherically blunted, 12.84 deg/7 deg bent nose biconic at Mach 6 in air were measured. The angle of attack, referenced to the axis of aft cone, was varied from 0 deg to 25 deg in nominal 5 deg increments. Two values of free stream Reynolds number based on model length were tested. Predictions from simple theories and from a supersonic, three dimensional, external inviscid code (STEIN) are compared with measured values. Predicted STEIN shock shapes and windward pressures are in agreement with measured values for both bionics over the present range of angle of attack.

213. **Miller, C. G. III; and Gnoffo, P. A.: Pressure Distributions and Shock Shapes for a Bent-Nose Biconic at Incidence. AIAA Journal, vol. 20, pp. August 1982, pp. 1150-1151.**

The purpose of this Note is to present the initial results of this experimental study and make comparisons to prediction. These results include pressure distributions and shock shapes measured on a spherically blunted bent-nose biconic model at Mach 6 in air for angles of attack from 0° to 25°.

214. **Miller, C. G. III; and Gnoffo, P. A.: An Experimental Investigation of Hypersonic Flow Over Bionics at Incidence and Comparison to Prediction. AIAA Paper 82-1382, August 1982.**

Pressure distributions, aerodynamic coefficients, shock shapes, and oil-flow patterns were measured on spherically blunted, 12.84 deg/7 deg on-axis and bent bionics at Mach 6 in air. The angle of attack, referenced to the axis of the aft-cone, was varied from -10° to 40° in nominal 5° increments. Predictions from an inviscid flowfield computer code referred to as STEIN and codes which solve the three-dimensional parabolized Navier Stokes (PNS) equations were compared with measurement. Three PNS codes were found which provided accurate predictions of windward and leeward pressure distributions for angles of attack to 25°.

215. **Miller, C. G.; Blackstock, T. A.; Helms, V. T.; and Midden, R. E.: An Experimental Investigation of Control Surface Effectiveness and Real-Gas Simulation for Biconics. AIAA Paper 83-0213, January 1983.**

Aerodynamic coefficients, pressure distributions, and oil-flow patterns measured on spherically blunted, 13 deg/7 deg on-axis and bent biconics at Mach 6 and 10 in air are compared. Angle of attack, referenced to the axis of the aft cone, was varied from 0° to 40°. The effect of deflection of base-mounted flaps on aerodynamic characteristics was examined at Mach 10. Real-gas effects on aerodynamic coefficients, pressure distributions and shock detachment distance were simulated by testing the biconic models in Mach 6 air and Mach 6 CF4 flows. (Density ratio equal to 5.3 for air and 12 for CF4). The on-axis biconic is stable but cannot be trimmed at angles of attack above zero with the assumed center of gravity and flap configuration. Flaps on the bent biconic produced as table and trim configuration for angles of attack from 0° to 23°. A significant effect of density ratio on aerodynamic coefficients, pressure distribution, and shock detachment distance was observed.

216. **Bryan, M. D.; Calloway, R. L.; and Blackstock, T. A.: Fine-Cut Bluntness Effects on the Hypersonic Stability of a 10 Degree Cone. AIAA Paper 84-0503, January 1984.**

A systematic wind-tunnel investigation has been performed on a series of spherically blunted 10° cones to determine Mach number and angle-of-attack effects on the hypersonic static stability. The cones, which range in nose-to-base radius ratios from 0 to 0.5, have been tested from -4° to 20° angle of attack at Mach numbers of 6 and 10 in air and 20 in helium. Relatively large excursions in the center-of-pressure location were observed for small changes in cone nose bluntness. The movement of the center of pressure was also noted to become more pronounced as Mach number increased; increases in angle of attack tended to lessen the magnitude of these excursions. Computational predictions of basic aerodynamic coefficients as well as the center-of-pressure locations using both viscous and inviscid theory compare very well with the experimental data.

217. **Miller, C. G.; Gnoffo, P. A.; and Wilder, S. E.: Experimental Heating Distributions for Biconics at Incidence in Mach 10 Air and Comparison to Prediction. AIAA Paper 84-2119, 1984.**

218. **Morrisette, E. L.; Creel, T. R.; and Chen, F. J.: Effects of Cone Surface Waviness on Transition in Quiet and Noisy Supersonic Freestreams. AIAA Paper 86-1086, May 1986.**

A comparison of transition on wavy-wall and smooth-wall cones in a Mach 3.5 wind tunnel is made under conditions of either low freestream noise (quietflow) or high freestream noise (noisy flow). The noisy flow compares to that found in conventional wind tunnels while the quiet flow gives transitional Reynolds numbers on smooth sharp cones comparable to those found in flight. The waves were found to have a much smaller effect on transition than similar sized trip wires. A satisfactory correlating parameter for the effect of waves on transition was simply the wave height-to-length ratio. A given value of this ratio was found to cause the same percentage change in transition location in quiet and noisy flows.

SUBJECT CATEGORY 2 - EXPERIMENTAL RESULTS - CONFIGURATIONS

1. **Penland, Jim A.; Ridyard, Herbert W.; and Fetterman, David E.: Lift, Drag, and Static Longitudinal Stability Data From an Exploratory Investigation at a Mach Number of 6.86 of an Airplane Configuration Having a Wing of Trapezoidal Plan Form. NACA RM L54L03b, January 1955.**

An investigation to determine the lift, drag, and static longitudinal stability characteristics of an airplane configuration having a wing of trapezoidal planform with modified hexagonal airfoil section, a sweep of 29° at the quarter chord line, a taper ratio of 0.140, an aspect ratio of 3.00 and 5° semiangle wedge tail sections has been made at a Mach number of 6.86 and Reynolds numbers of 343,000 and 566,000 based on wing mean aerodynamic chord. Data were obtained for angles of attack up to about 28° for the complete airplane configuration and up to about 14° for the body-alone, body-wing, and body-tail configurations.

2. **Ridyard, Herbert W.; Fetterman, David E.; and Penland, Jim A.: Static Lateral Stability Data From an Exploratory Investigation at a Mach Number of 6.86 of an Airplane Configuration Having a Wing of Trapezoidal Planform. NACA RM L55A21a, February 1955.**

An investigation to determine the static lateral stability characteristics of an airplane configuration having a wing of trapezoidal planform with modified hexagonal airfoil section, 29° sweepback at the quarter-chord line, a taper ratio of 0.140, an aspect ratio of 3.00, and tail surfaces with 5° semiangle wedge sections has been made at a Mach number of 6.86 and a Reynolds number of 343,000 based on wing mean aerodynamic chord. Data were obtained for angles of sideslip up to 10° and angles of attack up to 25° for the complete model and for other combinations of its components. The data are presented with respect to the body axes.

3. **Fetterman, David E.; Penland, Jim A.; and Ridyard, Herbert W.: Static Longitudinal and Lateral Stability and Control Data From an Exploratory Investigation of a Mach Number of 6.86 of an Airplane Configuration Having a Wing of Trapezoidal Planform. NACA RM L55C04, April 1955.**

An investigation to determine the static longitudinal and lateral stability and control characteristics of an airplane configuration having a wing of trapezoidal planform with modified hexagonal airfoil section, 29° sweepback at the quarter-chord line, a taper ratio of 0.140, an aspect ratio of 3.00, and tail surfaces with 5° semiangle wedge sections has been made at a Mach number of 6.86 and a Reynolds number of 343,000 based on wing mean aerodynamic chord. Data were obtained for angles of sideslip from -2° to 8° and angles of attack from -5° to 25° for the complete model with various combinations of its tail surfaces. The horizontal-tail incidence was varied from -20° to 2° and the vertical-tail incidence was varied from -6° to 6° .

4. **Penland, Jim A.; Fetterman, David E.; and Ridyard, Herbert W.: Static Longitudinal and Lateral Stability and Control Characteristics of an Airplane Configuration Having a Wing of Trapezoidal Planform with Various Tail Airfoil Sections and Tail Arrangements at a Mach Number of 6.86. NACA RM L55F17, August 1955.**

An investigation has been carried out in the Langley 11-inch hypersonic tunnel to determine the static longitudinal and lateral stability and control characteristics of an airplane configuration having a trapezoidal wing with a modified hexagonal airfoil section and equipped with various tail airfoil sections and tail arrangements. Tail airfoil sections tested were a 10° wedge, a flat-plate section, and a series of composite airfoils consisting of flat plates forward of the hinge lines and wedges behind the hinge lines. The tests were made at a Mach number of 6.86 and a Reynolds number of 343,000 based on the wing mean aerodynamic chord. Data were obtained for angles of sideslip up to 10° and angles of attack up to 25° for the complete model with the cruciform 10° wedge horizontal and vertical tails and for the complete model with various tail arrangements.

5. **Rainey, Robert W.: Static Stability and Control of Hypersonic Gliders.(U) NACA RM L58E12a, July 7, 1958.**

A study has been made of several hypersonic boost gliders, two of which are high-lift-drag-ratio types and are envisioned for operation within the sensible atmosphere and two which are low-lift-drag-ratio types which might be considered as orbital reentry vehicles. It appeared that, in general, it was possible to obtain the desired trim features for both types of vehicles. Furthermore, at the low angles of attack, static directional stability and control are adequate for the complete configurations investigated. Some problem areas are pointed out.

6. **Penland, Jim A. and Fetterman, David E.: High-Speed Static Stability Characteristics of the X-15 Airplane. USAF-USN-NACA Conference on Progress of X-15 Project, July 28-29, 1958, Los Angeles, California.**
7. **Feller, W. B. and Burbank, P. B.: Wind Tunnel Investigation of the Heat Transfer to the X-15 Airplane. Conference on Progress of the X-15 Project, July 1958.**

8. **Robinson, Ross B. and Bernot, Peter T.: Aerodynamic Characteristics at a Mach Number of 6.8 of Two Hypersonic Missile Configurations, One with Low-Aspect-Ratio Cruciform Fins and Trailing-Edge Flaps and One with a Flared Afterbody and All-Movable Controls. NACA RM L58D24, August 1958.**

The cruciform-fin configuration had triangular fins with an apex angle of 5° and inline, cruciform, rectangular, trailing-edge flaps. The model with all-movable controls consisted of a body with a 10° flared afterbody equipped with cruciform modified 70° delta controls mounted at the 46.7 percent body station. Tests were made through an angle-of-attack range of -2° to 20° to zero sideslip. Most of the tests were made to determine the effectiveness of both types of surfaces as longitudinal controls, but a limited amount of lateral-control data was obtained.

9. **Fetterman, David E., Jr.: Effects of Simulated Rocket-Jet Exhaust on Stability and Control of a Research-Type Airplane Configuration at a Mach Number of 6.86. NASA TM X-127, October 1959.**

An investigation has been made in the Langley 11-inch hypersonic tunnel at a free-stream Mach number of 6.86 to determine the jet-interference effects at high jet-static-pressure ratios on the stability and control of a research-type airplane configuration over an angle-of-attack range of $\pm 4^\circ$. The jet-interference flow field was defined from schlieren photographs of compressed-air tests conducted over a Reynolds number range of from 0.57×10^6 to 3.95×10^6 , based on fuselage length and a jet-static-pressure-ratio range of from 0 to 1460.

10. **Fetterman, David E.; and Penland, Jim A.: Static Longitudinal, Directional, and Lateral Stability and Control Data from an Investigation at a Mach Number of 6.83 of Two Developmental X-15 Airplane Configurations. NASA TM X-209, March 1960.**

11. **Penland, Jim A.; and Fetterman, David E.: Static Longitudinal, Directional, and Lateral Stability and Control at a Mach Number of 6.83 of the Final Configuration of the X-15 Research Airplane. NASA TM X-236, April 1960.**

An investigation of the final configuration 3 has been carried out. Data were obtained for angles of sideslip of 0° and -4.5° and angles of attack from -20° to 24° . The horizontal-tail deflection was varied from -35° to 15° , the vertical-tail deflection from 0° to -7.5° , and the speed-brake deflection from 0° to 50° .

12. **Byron, M. Jaquet: Static Longitudinal and Lateral Stability Characteristics at a Mach Number of 3.11 of Square and Circular Planform Reentry Vehicles, with Some Effects of Controls and Leading-Edge Extensions. NASA TM X-272, May 1960.**

Results are presented of an investigation to determine the static longitudinal and lateral stability and some control characteristics of a square planform reentry vehicle with leading-edge extensions and a circular planform reentry vehicle with pyramidal controls. Tests were made at Reynolds numbers per foot of $12,050 \times 10^6$ to $13,060 \times 10^6$ and at angles of attack from approximately -5° to 13° at two sideslip angles, 0° and 6° .

13. **Rainey, Robert W.; Fetterman, David E.; and Smith Robert: Summary of the Static Stability and Control Results of a Hypersonic Glider Investigation. NASA TM X-277, May 1960.**

An investigation has been made of the static stability and control characteristics of several versions of a high lift-drag flat-bottom hypersonic glider primarily at hypersonic speeds from Mach numbers of 6.7 to 18.4. The final configuration consisted of a delta wing with leading-edge sweep of 78° , a half-cone-cylinder body, toed-in wing-tip fins and a deflected nose. In general, satisfactory static stability and control characteristics were obtained.

14. **Ladson, Charles L.; Johnston, Patrick J.; and Trescot, Charlie D., Jr.: Effects of Wing Planform Geometry on the Aerodynamic Characteristics of a Hypersonic Glider at Mach Numbers up to 9.6. NASA TM X-286, May 1960.**

Tests were conducted on a winged, lifting hypersonic glider configuration to study the effects of wing crank and wing longitudinal location on the performance and static stability characteristics of such a vehicle throughout the Mach number range. Data were obtained at a Mach number of 0.92 in the Langley transonic blowdown tunnel, at Mach numbers of 1.62 and 2.91 in the Langley 9-inch tunnel, and at Mach numbers of 6.8 and 9.6 in the Langley 11-inch hypersonic tunnel at angles of attack up to about 25° . Includes data obtained in LaRC free-flight tunnel at 60'/sec on same configuration.

15. **Rainey, Robert W.; and Close, William H.: Studies of Stability and Control of Winged Reentry Configurations. (U) NASA TM X-327, September 1960.**

A study of several of the static stability and control problem areas of winged reentry vehicles capable of maximum lift-drag ratios of about 2 at hypersonic speeds. This study was carried out at speeds from subsonic to a Mach number of 18 and at angles of attack from 0° to 55° (approximately that for maximum lift). Although the magnitudes of the problems for specific vehicles would undoubtedly be altered from those contained within this generalized discussion, the principles behind which the solutions to those problems were reached should be applicable.

16. **Ladson, Charles L.; and Johnston, Patrick J.: Aerodynamic Characteristics of a Blunt-Nosed Winged Reentry Vehicle at Supersonic and Hypersonic Speeds. NASA TM X-357, February 1961.**

Performance, stability, and control data are presented at Mach numbers of 1.62 and 2.91 at angles of attack up to 15° and at Mach numbers of 6.8 and 9.6 at angles of attack up to 25° .

17. **Johnston, Patrick J; and Ladson, Charles L.: Experimental Investigation of Longitudinal Static Stability of a Glider and 11 Booster Configurations at Mach Numbers from 1.62 to 6.86. NASA TM X-356, March 1961.**

Reynolds numbers varied from 0.27×10^6 to 1.0×10^6 per inch. Results showed that the aerodynamic instability incurred as a result of rigidly mounting the glider in tandem with the boosters could be overcome by using fins on the booster but that these fins become ineffective at hypersonic speeds. Mounting the glider alongside of the boosters produced an aerodynamically stable configuration but the nonlinear and hysteresis effects in the force and moment results combined with high drag make it a less desirable configuration.

18. **Ladson, Charles H.: Directional and Lateral Stability Characteristics of a Winged Reentry Vehicle at Hypersonic Speeds. NASA TM X-550, August 1961.**

An investigation has been carried out at high sideslip angles at a Mach number of 9.6 and at high angles of attack (up to a maximum lift coefficient) at a Mach number of 6.7. The effects of rolling the tip fins out from the vertical were also studied at the higher angles of attack.

19. **Ladson, Charles H.: Effects of Several Nose and Vertical-Fin Modifications on the Low-Angle-of-Attack Static Stability of Winged Reentry Vehicle at Mach Numbers of 9.6 in Air and 17.8 in Helium. NASA TM X-608, November 1961.**

Longitudinal performance and stability characteristics are presented at angles of attack from -5° to 25° while directional and lateral aerodynamic characteristics are presented only for angles of attack of 0° and 10° .

20. **Hopkins, E. J.; Fetterman, D. E.; and Saltzman, E. J.: A Comparison of Full-Scale X-15 Lift and Drag Characteristics with Wind Tunnel Results and Theory. Research-Airplane-Committee Report on Conference on the Progress of the X-15 Project, November 20-21, 1961.**

Data on the lift and drag due to lift characteristics of the X-15 airplane obtained in flight are shown to be in agreement with wing-tunnel-model data for Mach numbers up to 5. Existing theoretical methods are indicated to be adequate for estimating the X-15 minimum drag but underestimated the drag due to lift and overestimated the maximum lift-drag ratio. Two-dimensional theory is shown to be adequate for predicting the base pressures behind surfaces having very blunt trailing edges, such as those on the vertical tail of the X-15.

21. **Ladson, Charles L.: Effects of Booster Transition Section and Rudder Deflection on the Low-Angle-of-Attack Static Stability of a Winged Reentry Vehicle at Mach Numbers of 10.8 and 17.8 in Helium (U) NASA TM X-624, January 1962.**

The tests were conducted in the Langley 11-inch hypersonic tunnel at Mach numbers of 10.8 and 17.8 with helium as the test medium. Longitudinal stability and performance characteristics are presented at angles of attack from -5° to 15° . Directional-control characteristics are presented only for an angle of attack of 0° for sideslip angles from -5° to 10° . No detailed analysis of the data is presented.

22. **Hopkins, Edward J.; Fetterman, David E., Jr.; and Saltzman, Edwin J.: Comparison of Full-Scale Lift and Drag Characteristics of the X-15 Airplane with Wind-Tunnel Results and Theory. (U) NASA TM X-713, March 1962.**

Comparisons are made between the minimum drag characteristics of the full-scale X-15 airplane and wind-tunnel model data and theory extrapolated to flight Reynolds number for Mach numbers of 2.5 and 3.0. Similar comparisons are made for drag due to lift and maximum lift-drag ratio for Mach numbers up to about 5. Speed-brake drag and base-drag results are presented up to Mach numbers of 5.5 and 6, respectively.

23. **Penland, Jim A.: A Study of the Aerodynamic Characteristics of a Fixed Geometry Paraglider Configuration and Three Canopies with Simulated Variable Canopy Inflation at a Mach Number of 6.6. NASA TN D-1022, March 1962.**

Three-component force tests have been made at a Reynolds number of 0.24×10^6 based on the keel length of a fixed-geometry paraglider configuration consisting of a canopy having leading-edge and shroud-line diameters of 1.8 percent of the keel length and a payload diameter of 11 percent of the keel length. Further force tests were made on unshrouded canopies with three different degrees of simulated canopy inflation. Oil-flow and temperature-sensitive-paint tests were made to study the effects of canopy geometry and shroud interference on local flow fields and aerodynamic heating. Measured force characteristics are compared with Newtonian theory.

24. **Whitcomb, Charles F.; Armstrong, William O.; and Swihart, John M.: Longitudinal and Lateral Aerodynamic and Stability Characteristics of a Model of a Preliminary Design Hydrogen-Fuel Hypersonic Delta-Wing Vehicle at Mach Numbers of 6.86 and 10.03. (U) NASA TM X-671, April 1962.**

An investigation was conducted in the Langley 11-inch hypersonic tunnel and the 15-inch hypersonic flow apparatus at Reynolds numbers of 1.48×10^6 and 0.74×10^6 . This study was made to determine the longitudinal, lateral, and directional stability characteristics of a model of a large-volume boost-to-cruise vehicle designed to fly above a Mach number of 7. Aerodynamic coefficients are presented for angles of attack from -3° to 11° and angles of sideslip from -2° to 10° . Theoretical estimates of the aerodynamic characteristics at a Mach number of 6.86 are also included.

25. **Johnston, Patrick J.; and Snyder, Curtis D.: Static Longitudinal Stability and Performance of Several Ballistic Spacecraft Configurations in Helium at a Mach Number of 24.5. NASA TN D-1379, August 1962.**

26. **Bertram, M. H.; Boswinkle, R. W., Jr.; Campbell, J. P.; Cohen, N. B.; Fetterman, D. E., Jr.; Grant, F. C.; Henry, J. R.; and Trimpf, R. L.: Aerodynamics of Space Vehicles. NASA SP 23, NASA-University Conference on the Science and Technology of Space Exploration, November 1962.**

The problems of efficient hypersonic flight of boost-glide and air-breathing vehicles are reviewed for the areas of aerodynamics, stability and control, heating, and air ingestion. The application of classical hypersonic solutions to this class of vehicles is shown although these solutions cannot always be applied without extensive modification. Particular attention is given to the problems of interference between major vehicle components and the internal flow problems of air breathers.

27. **Boswinkle, Robert W., Jr.: Aerodynamic Problems of Launch Vehicles. NASA SP 23, NASA-University Conference on the Science and Technology of Space Exploration, November 1962.**

The airflow about a launch vehicle causes problems which may affect the entire vehicle or may affect only localized areas; the problems can occur when the vehicle is on the launcher as well as during flight. Specific problems discussed include local steady-state loads, overall steady-state loads, buffet, ground wind loads, base heating, and rocket-nozzle hinge moments.

28. **Campbell, John P.: Low-Speed Aerodynamic Research Related to the Landing of Space Vehicles. NASA SP 23, NASA-University Conference on the Science and Technology of Space Exploration, November 1962.**

Aerodynamic research results related to the low-speed approach and landing of space vehicles are presented. Two basic types of space vehicles are covered: those designed for vertical or near-vertical descent (which require some auxiliary device for landing) and those which perform glide landings. Spacecraft discussed include nonlifting bodies (such as the Mercury spacecraft), low lift-drag-ratio lifting bodies, fixed-geometry glide-landing types similar to the Dyna-Soar, and variable-geometry glide-landing types which involve a change in configuration between reentry and landing. Aerodynamic characteristics are also presented for auxiliary landing aids which may replace the parachute for some recovery applications. These landing aids include the steerable parachute, the rotorchute, and the parawing which is being developed for use in Gemini recovery system.

29. **Baradell, Donald L.: Lateral Range Control by Banking During Initial Phases of Supercircular Reentries. NASA TN D-1511, November 1962.**

30. **Ladson, Charles H.: Summary of Aerodynamic Characteristics of a Winged Reentry Vehicle at Mach Numbers From 0.06 to 9.6 with Detailed Results at a Mach Number of 9.6. NASA TM X-765, March 1963.**

Tests have been conducted at the National Aeronautics and Space Administration, Langley Research Center to determine the effects of Mach number on some of the more important longitudinal, directional, and lateral stability and control parameters of a winged reentry vehicle. Detailed results obtained at a Mach number of 9.6 and an analysis of these results are presented, as well as a summary of data obtained at subsonic and supersonic speeds.

31. **Spencer, Bernard, Jr.; Henry, Beverly Z.; and Putnam, Lawrence E.: The Transonic Longitudinal and Lateral Aerodynamic Characteristics of a Low-Fineness Ratio Elliptic Hypersonic Configuration Employing Variable-Sweep Wing Panels for Improving Subsonic Lift and Performance. NASA TM X-768, 1963.**

32. **Spencer, Bernard, Jr.: Longitudinal Aerodynamic Characteristics at Mach Numbers From 0.40 to 1.10 of a Blunted Right-Triangular Pyramidal Lifting Reentry Configuration Employing Variable-Sweep Wing Panels. NASA TN D-1518, March 1963.**

An investigation has been conducted in the Langley 7- by 10-foot transonic tunnel at Mach numbers from 0.40 to 1.10 on a right-triangular pyramidal-type reentry configuration employing variable wing sweep as a means of increasing lift and lift-drag ratio at subsonic speeds. Various means of longitudinal control were also tested in conjunction with the range of wing-panel sweeps at a Mach number of 0.40.

33. **Johnston, Patrick J.: Longitudinal Aerodynamic Characteristics of Several Fifth-Stage Scout Reentry Vehicles from Mach Number 0.60 to 24.4 Including Some Reynolds Number Effects on Stability at Hypersonic Speeds. NASA TN D-1638, May 1963.**

34. **Spencer, Bernard, Jr.; and Phillips, W. Pelham: Effects of Cross-Section Shape on the Low-Speed Aerodynamic Characteristics of a Low-Wave-Drag Hypersonic Body. NASA TN D-1963, October 1963.**

The effects on the longitudinal and lateral aerodynamic characteristics of changing body cross-sectional shape from circular to elliptic on a body having a fineness ratio of 10.00 are presented for low subsonic speeds. Also included are the effects of positive and negative camber for the circular body and for some of the elliptic bodies.

35. **Becker, John V.: Entry Vehicles. NASA RP-118, pp. 70-74, November 1963.**

In the development of entry vehicle designs, a great amount of research is centered on the choice of hypersonic maneuverability or lift-drag ratio. Different configurations are discussed in an attempt to select the design having the most desirable lift-drag characteristics. The first lifting body to be investigated extensively is based on the round-bottomed half-cone shape with the aim of deriving some advantage from available cone theory. The modified flat-bottomed half-cone vehicle and other thickened delta wing designs with various degrees of nose blunting, thickness, and camber distributions have also appeared. It is made clear that landable entry vehicles comprise a complex synthesis of advanced research data from many disciplines, blended together with many assumptions and compromises representing the arbitrary preferences of the designer.

36. **Spencer, Bernard, Jr.; and McShera, John T., Jr.: Longitudinal and Lateral Aerodynamic Characteristics at Mach Numbers of 3.00, 3.96, and 4.65 of a Low-Fineness-Ratio Elliptical Hypersonic Configuration Having Variable-Sweep Wing Panels. NASA TM X-807, 1963.**
37. **Rainey, Ruby W.; and Ladson, Charles L.: Preliminary Aerodynamic Characteristics of a Manned Lifting Entry Vehicle at a Mach Number of 6.8. NASA TM X-844, August 1963.**
38. **Ladson, Charles L.: Aerodynamic Characteristics of a Manned Lifting Entry Vehicle at a Mach Number of 6.8. NASA TM X-915, February 1964.**
39. **Arrington, James P.; and Maddalon, Dal V.: Aerodynamic Characteristics of Several Lifting and Nonlifting Configurations at Hypersonic Speeds in Air and Helium. NASA TM X-918, June 1964.**
40. **Goetz, Robert C.: Effects of Leading-Edge Sweep on Flutter Characteristics of Some Delta-Planform Surfaces at a Mach Number of 15.4. NASA TN D-2360, July 1964.**

A wind-tunnel investigation was conducted in helium flow at Mach 15.4 to determine some effects of leading-edge sweep and profile shape on the flutter characteristics of some delta-planform all movable control surfaces. The profile shapes tested were blunt leading-edge wedges, double wedges, and slabs. In general, the results indicate that increasing the leading-edge-sweep angle from 60° to about 65° or 70° is destabilizing, while further increases in sweep are stabilizing. However, these trends may be influenced by the layer of disturbed flow along the reflection plane surface, particularly for the more highly swept models.

41. **Rainey, Ruby W.; and Ladson, Charles, L.: Aerodynamic Characteristics of a Manned Lifting Entry Vehicle at $M = 0.2$ and $M = 1.2$. NASA TM X-1015, September 1964.**

42. **Becker, J. V.: Studies of High Lift/Drag Ratio Hypersonic Configurations. ICAS Paper 64-551, 1965.**

The comparative performance and heat-transfer problems of a variety of simple lifting forms are investigated in the Mach range from 5 to 20. In general it is found that the use of flat lifting surfaces produces higher lift/drag ratios than circular, vee, "caret", and "isentropic compression" surfaces. An exception is the thin delta wing with underslung body for which a favorable interference effect is produced at the lower hypersonic speeds. Configuration limits within which the favorable interference is obtainable are established. As the Mach number advances beyond about 10 adverse viscous interactions develop to the extent that the interference effect becomes unfavorable, and higher lift/drag ratios are produced by flat-bottom wing-bodies. A comparison of the best of these wing-body combinations with thick wings and with merged wing-bodies of equal volume and planform area reveals only small differences in peak lift/drag ratio. All configurations suffer a major deterioration in lift/drag as the Mach number is increased and the Reynolds number is decreased in conformance to typical hypersonic flight trajectories, because of the growth of adverse viscous effects. An important factor in this performance degradation is the unexpectedly high pressure found on the lee side at the higher Mach numbers. Heat-transfer studies reveal several anomalies associated with the viscous interactions and with the intersecting flow fields of wing and body. Analysis of the leading-edge region of a typical wing leads to the conclusion that the relatively sharp leading edges essential to high lift/drag ratio are not impractical if a combination of internal and radiative cooling techniques are employed.

43. **Boyd, J. M.: Optimal Utilization of Supersonic Favorable Interference to Obtain High Lift-Drag Ratios. AIAA Paper 65-752, 1965.**

44. **Fetterman, D. E.: Favorable Interference Effects on Maximum Lift-Drag Ratios at Half-Cone Delta-Wing Configurations at Mach 6.86. NASA TN D-2942, August 1965.**

The characteristics of half-cone delta-wing configurations is investigated under predominantly laminar boundary-layer conditions at a Mach number of 6.86 and Reynolds number based on model length of 1.43×10^6 to determine the availability of favorable interference effects for improving the maximum lift-drag ratio $(L/D)_{max}$. Simple modification to the half-cone body to provide more volume and better volume distribution are also considered. Approximate solutions for the characteristics of half-cone winged configurations are included in the appendixes.

45. **Spencer, Bernard, Jr.; and Ray, Edward J.: Transonic Longitudinal Aerodynamic Characteristics of an Airbreathing Reusable Booster Model Having Variable-Sweep Wings. NASA TM X-1093, 1965.**

46. **Bertram, M. H.; Fetterman, D. E.; Henderson, A., Jr.; Johnston, P. J.: Attainment of High Lift-Drag Ratios at Hypersonic Speeds. NASA TN D-2956, August 1965.**

An evaluation of results of recent investigations of slender configurations to determine factors having the most significant influence on aerodynamic efficiency is presented. The results, which were obtained at a Mach number of 6.8 in air and of 20 in helium, show the effects on maximum lift-drag ratio of viscosity body longitudinal curvature, cross-sectional shape, and fineness ratio, wing location, planform leading-edge sweep and diameter, and volumetric efficiency. In addition, the interrelationship of these factors in determining the extent of beneficial effects from favorable interference is examined. Included also are preliminary comparisons between certain merged wing-body configurations and discrete wing-body types.

47. **Johnston, Patrick J.: Stability Characteristics of a Manned Lifting Entry Vehicle at a Mach Number of 20 in Helium. NASA TM X-1156, August 1965.**
48. **Rainey, Robert W.: Summary of an Advanced Manned Lifting Entry Vehicle Study. NASA TM X-1159, 1965.**
49. **Trescot, Charles D., Jr.; and Spencer, Bernard, Jr.: Hypersonic Aerodynamic Characteristics of a Lifting Reentry Vehicle Model With Four Types of Longitudinal Control Surfaces. NASA TM X-1173, November 1965.**

An investigation has been made in the Langley 15-inch hypersonic flow apparatus at a Mach number of 10.03 to determine the effects of four types of longitudinal controls on the static longitudinal, lateral, and directional stability characteristics of a reentry vehicle configuration. The four types of controls tested were aft mounted fins, a canard, a chin flap, and two body trailing-edge flaps. The angle-of-attack range was varied from about -8° to 41° at a sideslip angle of 0° and from about -8° to 20° at a sideslip angle of -5° . The Reynolds number, based on the body reference length of 10.00 inches (0.254 meter), was 1.40×10^6 .

50. **Whitehead, A. H., Jr.: Effect of Body Cross Section and Width-Height Ratio on Performance of Bodies and Delta-Wing Body Combinations at Mach 6.9. NASA TN 2886, June 1966.**

An investigation of the aerodynamic characteristics of bodies and wing-body combinations with triangular, rectangular, and elliptical body cross-sectional shapes and with body width-height ratios of 2 and 3 was conducted at a free-stream Mach number of 6.9 and a Reynolds number based on length of 1.4×10^6 . The two delta wings tested in combination with these bodies had leading-edge sweep angles of 70° and 75° . All configurations were tested in both flat-top and flat-bottom orientations.

51. **Saltzman, E. J.; and Garringer, D. J.: Summary of Full-Scale Lift and Drag Characteristics of the X-15 Airplane. NASA TN D-3343, March 1966.**

52. **Brooks, Cuyler W., Jr.; and Cone, Clarence D., Jr.: Hypersonic Aerodynamic Characteristics of Aircraft Configurations with Canard Controls. NASA TN D-3374, April 1966.**

The aerodynamic characteristics of a canard-control, wing-body configuration having a 70°-swept-delta wing were studied at a Mach number of 10.03 in hypersonic flow apparatus. The effects of canard size and planform, body length, wing position, and vertical-tail position on the longitudinal, lateral, and directional characteristics were determined. The stability and lift characteristics of these configurations are nonlinear because of the nature of the hypersonic flow regime; these experimental nonlinearities are qualitatively predicted by Newtonian impact theory. Effects of canard deflection and wing vertical position on the variation of lift coefficient with angle of attack indicate that there is a measurable interference effect of the canard shock field and/or wake on the flow under the high wing. Canard control effectiveness is somewhat greater for the trapezoidal than for the delta canard and increases with body length (canard moment arm) for the high-wing configurations. For the low-wing configurations an increase is found only for the highest canard deflection. The canards have negligible effects on the lateral and directional stability of the configurations except at the higher angles of attack. The vertical tails are directionally stabilizing, as would be expected, but the dihedral effect of the vertical tails is dependent on the wing vertical position. The high-wing configurations are considerably more stable laterally and directionally than the corresponding low-wing configurations.

53. **Fetterman, David E.; McLellan, Charles H.; Jackson, L. Robert; Henry, Beverly Z., Jr.; and Henry, John R.: A Review of Hypersonic Cruise Vehicles. NASA TM X-1276, October 1966.**

In an attempt to provide a focus for future aerodynamic programs in the development of hydrogen-fueled hypersonic cruise vehicles, the present status of the structural, propulsive, and aerodynamic research is examined to extract the presently known factors that significantly affect vehicles definition. Existing wing and body structural concepts and cryogenic-tankage thermal-protection systems are illustrated, possible inlet-engine arrangements are discussed, and the status of important local aerodynamic heating areas is briefly reviewed. In addition, uncertain areas which require further fundamental research and obstacles which hinder development are also pointed out. In general, existing structural and propulsive technologies for Mach 6 to 8 vehicles favor a discrete low-wing--body arrangement with a two dimensional inlet mounted in the wing pressure field. Aerodynamic considerations, however, indicated equal performance possibilities for either discrete wing-body or blended wing-body arrangements. The paper concludes with a discussion of several possible design concepts which conform to current guidelines and which are planned for future research.

54. **Ladson, C. L.: Effects of Various Canopies on the Aerodynamic Characteristics of a Manned Lifting Entry Vehicle at Mach 0.06 to 6.8. NASA TM X-1321, November 1966.**

The aerodynamic characteristics of a canard-control, wing-body configuration having a 70° swept-delta wing were studied at a Mach number of 10.03 in hypersonic flow apparatus. The effects of canard size and planform, body length, wing position, and vertical-tail position on the longitudinal, lateral, and directional characteristics were determined. The stability and lift characteristics of these configurations are nonlinear because of the nature of the hypersonic flow regime; these experimental nonlinearities are qualitatively predicted by Newtonian impact theory. Effects of canard deflection and wing vertical position and the variation of lift coefficient with angle of attack indicate that there is a measurable interference effect of the canard shock field and/or wake on the flow under the high wing. Canard control effectiveness is somewhat greater for the trapezoidal than for the delta canard and increases with body length (canard moment arm) for the high-wing configurations. For the low-wing configurations an increase is found only for the highest canard deflection. The canards have negligible effects on the lateral and directional stability of the configurations except at the higher angles of attack. The vertical tails are directionally stabilizing, as would be expected, but the dihedral effect of the vertical tails is dependent on the wing vertical position. The high-wing configurations are considerably more stable laterally and directionally than the corresponding low-wing configurations.

55. **Ashby, George C., Jr.; and Stone, David R.: Aerodynamic Effect of Volume Addition to a High L/D Wing-Body at Mach 6.0. Journal of Spacecraft and Rockets, Vol. 4, No. 2, February 1967.**
56. **Hefner, J. N.; and Keyes, J. W.: Effect of Forward-Facing Jets on Aerodynamic Characteristics of Blunt Configurations at Mach 6. Journal of Spacecraft and Rockets, vol. 4, April 1967, pp. 533-534.**
57. **Penland, J. A.; Edwards, C. L. W.; Witcofski, R. D.; and Marcum, D. C.: A Comparative Aerodynamic Study of Two Hypersonic Cruise Aircraft Configurations Derived from Trade-Off Studies. NASA TM X-1436, October 1967. (CONFIDENTIAL)**

Analytic and experimental evaluations are made of two hypersonic cruise configurations designed to common ground rules. The ground rules were established to represent commercial-transport vehicles with a payload of 200 passengers, a range of 5000 nautical miles, and a cruise speed of Mach 6. Liquid hydrogen fuel was necessary to meet the range requirement. The two evolving configuration concepts were a delta-wing vehicle with a distinct body and wing having elevators for longitudinal control and a blended wing body with double delta planform and elevons for pitch control. Both configurations were tested at Mach numbers from 0.1 to 1.2 and at 6.0. Tests of modifications to the delta-wing configuration were also made at Mach 6. Experimental and calculated results are presented for assessing longitudinal stability and control, directional stability, aerodynamic performance characteristics, and the effect of configuration modifications on maximum lift-drag ratio. The results of preliminary calculations are also included to show first-order effects of the nozzle exhaust on longitudinal trim.

58. **Brooks, Cuyler W., Jr.: Interference Effects of Canard Controls on the Longitudinal Aerodynamic Characteristics of a Winged Body at Mach 10. NASA TN D-4436, April 1968.**

An experimental investigation of the interference effects of canard controls on two hypersonic winged configurations was made at a Mach number of 10 in the Langley 15-inch hypersonic flow apparatus. The effect of variations in canard size and shape, body length, wing planform, and wing vertical position was determined. The results indicate that the canard control induces a broad pattern of interference in which the average flow angle over the wing and body surfaces downstream of the canard varies significantly from the free-stream value. The magnitude of the interference increases with increasing canard deflection and, in general, with increasing angle of attack. The most significant configuration parameters (aside from canard deflection) are wing position and canard size. The low-wing configurations are affected considerably less by interference than the high-wing configurations, especially at higher angles of attack. As would be expected, the larger canard causes larger disturbances in the flow than does the smaller canard.

59. **Andrews, Earl H., Jr.; and Rogers, R. Clayton: A Study of Underexpanded Exhaust Jets of an X-15 Airplane Model and Attached Ramjet Engine Simulator at M = 6.86. NASA TM X-1571, May 1968.**

An investigation was conducted using a 0.02-scale model of the X-15 airplane with a ramjet engine simulator attached to the underside of its afterbody. The investigation was initiated to approximate the effects of the ramjet exhaust plume upon the external pressures of a nozzle extension of an X-15 engine. The nozzle extension was tested with and without a simulated manifold designed to collect and to eject overboard auxiliary gases from the X-15 airplane.

60. **Langhans, Richard A.: Transonic Aerodynamic Characteristics of Two Hypersonic Cruise Configurations. NASA TM X-1816, June 1969.**

An investigation has been conducted in the Langley 8-foot transonic pressure tunnel to determine the aerodynamic characteristics of two hypersonic cruise airplane configurations. The investigation was conducted at Mach numbers ranging from 0.80 to 1.20, angles of attack from approximately -5° to 15° , and a constant Reynolds number per meter of 13.12×10^6 . The results of the investigation show the distinct wing-body configuration is longitudinally unstable for the chosen center-of-gravity location and has a pitch-up tendency. This configuration is directionally stable and has positive effective dihedral. The blended wing-body configuration is longitudinally stable at Mach numbers greater than 0.90 and unstable at Mach numbers from 0.80 to 0.90. This configuration is directionally stable and has positive effective dihedral.

61. **Spencer, Bernard, Jr.: Effects of Stabilizer Configuration on Transonic Aerodynamic Characteristics of a Variable-Geometry High-Hypersonic Performance Spacecraft. NASA TM X-1865, 1969.**

62. **Nelms, W. P., Jr.; Carmichael, R. L.; and Castellano, C. R.: An Experimental and Theoretical Investigation of a Symmetrical and a Cambered Delta Wing Configuration at Mach Numbers from 2.0 to 10.7. NASA TN D-5272, June 1969.**

63. **Goldberg, Theodore J.; Hefner, Jerrt N.; and Stone, David R.: Hypersonic Aerodynamic Characteristics of Two Delta-Wing X-15 Airplane Configurations. NASA TN D-5498, October 1969.**

An investigation of the hypersonic aerodynamic characteristics of two delta-wing X-15 research configurations was conducted at a Mach number of 6 and a Reynolds number of 2.71×10^7 per meter (8.27×10^6 per ft). Limited tests of one model were also made at a Mach number of 8. Results are presented to show the aerodynamic effects of wing geometry and longitudinal position, wing fins, nose cant, strakes, speed brakes, and a suspended test ramjet. The type of boundary layer ahead of the elevons was determined from oil-flow studies of the separation boundaries with and without boundary-layer trips. Experimental aerodynamic characteristics are compared with analytical estimates.

64. **Woods, W. C.; Arrington, J. P.; and Stone, D. R.: Hypersonic Performance Stability and Control of a Manned Space Shuttle Candidates. Presented at the 48th Annual Meeting of the Virginia Academy of Sciences, May 5-8, 1970, Richmond, Virginia (Langley Reprint L-7267).**

65. **Freeman, Delma C., Jr.; and Jones, Richard S.: Low-Speed Static Stability and Control Characteristics of Two Small-Scale, Hypersonic Cruise Configurations. NASA TM X-2021, June 1970.**

A wind-tunnel investigation has been conducted to determine the low-speed static stability and control characteristics of two small-scale, hypersonic cruise configurations - a distinct wing-body configuration and a blended wing-body configuration.

66. **Nelms, W. P., Jr.; and Axelson, J. A.: Effects of Wing Elevation, Incidence, and Camber on the Aerodynamic Characteristics of a Representative Hypersonic Cruise Configuration at Mach Numbers from 0.65 to 10.70. NASA TN D-6049, October 1970.**

67. **Ellison, James C.: Investigation of the Aerodynamic Characteristics of a Hypersonic Transport Model at Mach Numbers to 6. NASA TN 0-6191, April 1971.**

A wind-tunnel investigation was conducted at subsonic, supersonic, and hypersonic speeds to determine the aerodynamic characteristics of a blended wing-body configuration. Data were obtained for the configuration with a single vertical tail, a flow-through inlet, and elevons deflected from 5° to -20° . At a Mach number of 6 and a Reynolds number of 21.6×10^6 , the configuration had a maximum lift-drag ratio of 5.0 with no trim penalty. Based on the selected center-of-gravity position of 56.4 percent of the model length, the configuration was stable in all directions over the Mach number range. Results of analytical methods agreed with the data at zero elevon deflection for the range of angle of attack from 0° to 6° at all Mach numbers.

68. **Nelms, W. P., Jr.; and Axelson, J. A.: Longitudinal Aerodynamic Characteristics of Three Representative Hypersonic Cruise Configurations at Mach Numbers from 0.65 to 10.70. NASA TM X-2113, October 1970.**

69. **Bushnell, Dennis M.: Hypersonic Airplane Aerodynamic Technology. NASA SP-292, Conference on Vehicle Technology for Civil Aviation, the Seventies and Beyond, Paper No. 5, 1979.**
70. **Jernell, Lloyd S.: Aerodynamic Characteristics of Wing-Body and Lifting Body Configurations of Hypersonic Cruise Aircraft at Mach 2.30 to 4.63. NASA TM X-2287, July 1971.**

An investigation has been conducted to determine the effects of wing area and body-cross-section ellipticity ratio on the aerodynamic characteristics of a series of delta-planform wing-body configurations representing some of the principal features of a hypersonic cruise aircraft. The effects of body cross-section shape were also investigated for several lifting-body configurations. Data were obtained at angles of attack to approximately 24° for angles of sideslip of 0° and 3° . The Reynolds number was 9.8×10^6 per meter (3×10^6 per ft).

71. **Ellison, James C.; and Spencer, Bernard Jr.: Hypersonic Aerodynamic Characteristics of a Variable-Dihedral Delta-Wing Spacecraft Configuration. NASA TM X-2391, September 1971.**

An investigation of a variable-dihedral delta-wing spacecraft concept has been conducted in the Langley hypersonic flow apparatus at a Mach number of 10.03. Angle of attack was varied from -5° to about 38° at angles of sideslip of 0° and -5° . Static longitudinal and lateral-directional characteristics were investigated for wing dihedral angles of 0° , 30° , 60° , 90° , and 110° for both 0° and 13° of wing toe-in.

72. **Penland, J. A.; and Romeo, D. J.: Advances in Hypersonic Exploration Capability--Wind Tunnel to Flight Reynolds Number. Journal of Aircraft, vol. 8, no. 11, November 1971, pp. 881-884.**

In the past, the limited Reynolds number capability of hypersonic facilities has prevented reliable extrapolation of data to flight Reynolds numbers. Recent results on a hypersonic cruise aircraft configuration obtained at Mach 8 in the Cornell Aeronautical Laboratory Hypersonic Shock Tunnel over a Reynolds number range from a completely laminar boundary-layer to a predominately turbulent one are presented. The significant factors which can affect extrapolation of wing-tunnel data at subscale Reynolds numbers to flight values are identified. The capability for predicting turbulent flight Reynolds number data from wind-tunnel data under laminar and transitional boundary-layer conditions are shown.

73. **Nelms, Walter P., Jr.; and Thomas, C. L.: Aerodynamic Characteristics of an All-Body Hypersonic Aircraft Configuration at Mach Numbers from 0.65 to 10.6. NASA TN D-6577, November 1971.**

Aerodynamic characteristics of a model designed to represent an all-body, hypersonic cruise aircraft are presented for Mach numbers from 0.65 to 10.6. The configuration had a delta planform with an elliptic cone forebody and an afterbody of elliptic cross section. Detailed effects of varying angle of attack (-2° to $+15^\circ$), angle of sideslip (-2° to $+8^\circ$), Mach number, and configuration buildup were considered. In addition, the effectiveness of horizontal tail, vertical tail, and canard stabilizing and control surfaces was investigated. The results indicate that all configurations were longitudinally stable near maximum lift-drag ratio and the configurations with the vertical tails were directionally stable at all angles of attack. Trim penalties were small at the hypersonic speeds for a center-of-gravity location representative of the airplane, but because of the large rearward travel of the aerodynamic center, trim penalties were severe at transonic Mach numbers.

74. **Ashby, George C., Jr.: Experimental Boundary Layer Edge Mach Numbers for Two Space Shuttle Orbiters at Hypersonic Speeds. NASA TN D-6574, February 1972.**

Pitot-pressure profiles and surface pressure measurements have been obtained at four axial stations along the center lines of space shuttle straight-wing and delta-wing orbiters at angles of attack from 20° to approximately 60° and Mach numbers of 20.3 (helium) and 6.8 (air). The results show that the Mach number at the edge of the boundary layer is best predicted by tangent-cone theory up to shock detachment.

75. **Arrington, James P.; and Woods, William C.: Hypersonic Aerodynamic and Entry-Maneuver-Aerothermodynamic Interactions for Two Lifting Entry Vehicles. NASA TN D-6753, May 1972.**

The longitudinal, directional, and lateral static stability and control characteristics of a delta lifting body and a delta-wing body have been obtained at a Mach number of 20 in helium for operational Reynolds numbers over an angle-of-attack range of -4° to 55° . The aerodynamic characteristics of the wing body were then evaluated in an entry study to examine the effects of vehicle performance on the aerothermodynamic parameters associated with constant and variable angle-of-attack modes for a 1500-n.-mi. cross range.

76. **Nelms, Walter P., Jr.: Effects of Body Shape on the Aerodynamic Characteristics of an All-Body Hypersonic Aircraft Configuration at Mach Numbers From 0.65 to 10.6. NASA TN D-6821, May 1972.**

An experimental investigation was conducted to determine the effects of several variations in body shape on the aerodynamic characteristics of an all-body hypersonic aircraft configuration. The basic configuration had a delta planform with an elliptic cone forebody and an afterbody of elliptic cross section terminating in a straight-line trailing edge. Variations in body shape included the ratio of maximum cross-sectional to body planform area (0.0935 and 0.0625), body leading-edge sweep (75° and 80°), and forebody length ratio (0.667 and 0.750). In addition, the effects of a thin wing (1.5 percent thick) mounted on one of the bodies was investigated, and the aerodynamic characteristics of just the forebodies of two of the configurations were determined. The models had no stabilizing surfaces or propulsion system packages. Ranges of angle of attack (-4° to $+15^\circ$) and angle of sideslip (-4° to $+8^\circ$) were investigated. The results indicate that of the four complete bodies, the configuration with the lowest ratio of cross-sectional to body planform area had the highest maximum lift-drag ratio and the greatest level of longitudinal stability at most Mach numbers. All the configurations had positive longitudinal stability near maximum lift-drag ratio at most Mach numbers. With exception of the lowest subsonic Mach numbers, changes in body sweep angle and in forebody length ratio had only minor effects on maximum lift-drag ratio.

77. **Arrington, J. P.; and Ashby, G. C., Jr.: Effect of Configuration Modifications on the Hypersonic Aerodynamic Characteristics of a Blended Delta Wing-Body Entry Vehicle. NASA TM X-2611, September 1972.**

78. **Spencer, Bernard, Jr.; and Fournier, Roger H.: Supersonic Aerodynamic Characteristics of a Variable-Geometry Spacecraft Designed for High Hypersonic Performance. NASA TM X-2703, January 1973.**

An investigation has been made in the high Mach number test section of the Langley Unitary Plan wind tunnel on a variable-geometry high hypersonic performance spacecraft concept at Mach numbers from 2.30 to 4.63. The basic lifting body is designed for hypersonic lift-drag ratio near 3.0. The variable-geometry feature is a single-pivot two-position high wing which is deployed at subsonic speeds to improve vehicle landing characteristics. For the present investigation the wing was maintained in a stowed position, and the effects of horizontal stabilizer dihedral, elevon control effectiveness, and the addition of either a conventional single vertical tail or dorsal-fin-type vertical stabilizers on the longitudinal and lateral-directional stability and control characteristics were studied.

79. **Blair, A. B., Jr.; and Grow, Josephine: Aerodynamic Characteristics of a 55° Clipped-Delta-Wing Orbiter Model at Mach Numbers From 1.60 to 4.63. NASA TM X-2748, May 1973.**

An investigation has been conducted in the Langley Unitary Plan wind tunnel to determine the supersonic aerodynamic characteristics of a delta-wing orbiter model. The model was tested at Mach numbers from 1.60 to 4.63, at nominal angles of attack from -2° to 30° , at nominal angles of sideslip from -4° to 10° , and at Reynolds numbers from 5.9×10^6 to 8.2×10^6 per meter (1.8×10^6 to 2.5×10^6).

80. **Spencer, Bernard, Jr.; and Fournier, Roger H.: Supersonic Aerodynamic Characteristics of Hypersonic Low Wave Drag Elliptical Body-Tail Combinations as Affected by Changes in Stabilizer Configuration. NASA TM X-2747, July 1973.**

An investigation has been made at Mach numbers from 1.50 to 4.63 to determine systematically the effects of the addition and position of outboard stabilizers and vertical- and vee-tail configurations on the performance and stability characteristics of a low-wave-drag body as determined for the geometric constraints of length and volume. The elliptical cross section had an axis ratio of 2 (major axis horizontal) and an equivalent fineness ratio of 6.14. Base-mounted outboard stabilizers were tested at various dihedral angles from 90° to -90° with and without a single center-line vertical tail or a vee-tail. The angle of attack was varied from about -6° to 27° at sideslip angles of 0° and 5° and a constant Reynolds number of 4.58×10^6 (based on body length).

81. **Clark, Louis E.: Hypersonic Aerodynamic Characteristics of an All-Body Research Aircraft Configuration. NASA TN D-7358, December 1973.**

An experimental investigation was conducted at Mach 6 to determine the hypersonic aerodynamic characteristics of an all-body, delta-planform, hypersonic research aircraft (HYFAC configuration). The aerodynamic characteristics were obtained at Reynolds numbers based on model length of 2.84 million and 10.5 million and over an angle-of-attack range from minus 4 deg to 20 deg. The experimental results show that the HYFAC configuration is longitudinally stable and can be trimmed over the range of test conditions. The configuration had a small degree of directional stability over the angle-of-attack range and positive effective dihedral at angles of attack greater than 2 deg. Addition of canards caused a decrease in longitudinal stability and an increase in directional stability. Oil-flow studies revealed extensive areas of separated and vortex flow on the fuselage lee surface. A limited comparison of wind-tunnel data with several hypersonic approximations indicated that, except for the directional stability, the tangent-cone method gave adequate agreement at control settings between 5 deg and minus 5 deg and positive lift coefficient. A limited comparison indicated that the HYFAC configuration had greater longitudinal stability than an elliptical-cross-section configuration, but a lower maximum lift-drag ratio.

82. **Penland, Jim A.; Creel, Theodore R., Jr.; and Howard, Floyd G.: Experimental Low-Speed and Calculated High-Speed Aerodynamic Characteristics of a Hypersonic Research Airplane Concept Having a 65° Swept Delta Wing. NASA TN D-7633, August 1974.**

An experimental wind-tunnel investigation has been carried out to determine the static longitudinal, lateral, and directional stability and control characteristics of a model of a large-body, delta-wing hypersonic research airplane concept at low speed. This investigation was conducted at a dynamic pressure of 239.4 Pf (5 psf) and a Reynolds number, based on fuselage length, of 2×10^6 . The configuration variables included vertical fins, engine modules, canards, and a canopy. The aerodynamic results of a computer study at Mach numbers of 3 to 12 are presented.

83. **Stone, David R.; and Spencer, Bernard, Jr.: Aerodynamic and Flow Visualization Studies of Variations in the Geometry of Irregular Planform Wings at a Mach Number of 20.3. NASA TN D-7650, 1974.**

The longitudinal and summary lateral-directional stability characteristics have been obtained for a variety of irregular planform wings applied to a conceptual space shuttle orbiter. Three basic wing planforms with leading-edge sweep angles of 53.2 deg, 46.8 deg, and 35 deg were studied in conjunction with a series of inboard planform fillets with sweep angles up to 78 deg. The spanwise intersection point of the fillets and the basic wings was held constant. The data were obtained in the Langley 22-inch helium tunnel at a Mach number of 20.3 and a Reynolds number of 2.10 million based on model length. Model angle-of-attack range was from 0 deg to 54 deg at sideslip angles of 1 deg and minus 3.8 deg. Also included are results of a flow-visualization study consisting of electron-beam-illuminated flow and surface oil-flow patterns.

84. **Creel, Theodore R., Jr.; and Penland, Jim A.: Low-Speed Aerodynamic Characteristics of a Hypersonic Research Airplane Concept Having a 70° Swept Delta Wing. NASA TM X-71974, August 1974.**

An experimental investigation of the low-speed static longitudinal, lateral and directional stability characteristics of a hypersonic research tunnel with a 12-foot (3.66 meter) octagonal test section at the Langley Research Center. Aircraft component variations included: fuselage shape modifications, tip fins, center vertical fin, wing camber, and wing planform. This investigation was conducted at a dynamic pressure of 262.4 Pa (5.48 psf), a Mach number of 0.06, and a Reynolds number of 2.24×10^6 , based on body length. Tests were conducted through an angle-of-attack range of 0° to 30° with elevon deflections from +5.0° to -30.0°. The complete configuration exhibited positive static longitudinal, lateral, and directional stability up to angles of attack of at least 20° and was trimmable to lift coefficients of at least 0.70 with elevon deflections of -30°.

- 85. Johnson, Charles B.; and Marcum, Don C., Jr.: A Parametric Study of Effect of Forebody Shape on Flow Angularity at Mach 8. NASA TN D-7768, November 1974.**

Flow angularity and static pressure measurements have been made on the lower surface of nine forebody models that simulate the bottom forward surface of a hypersonic aircraft. Measurements were made in an area of the forebody that represents the location of an inlet of a scramjet engine. Tests were conducted at a Mach number of 8 for free-stream unit Reynolds numbers per meter of 28×10^6 for angles of attack of 0° , 5° , and 10° and 22×10^6 for angles of attack of 15° and 20° . A parametric variation of the forebody surface investigated the effect of: (1) spanwise curvature, (2) longitudinal curvature, and (3) planform shape on both flow angularity and static pressure distribution. Results of each of the three parametric variations of geometry were compared to those for the same flat delta forebody.

- 86. Kaufman, Louis G., II; and Johnson, Charles B.: Weak Incident Shock Interactions with Mach 8 Laminar Boundary Layers. NASA TN D-7835, December 1974.**

Weak shock-wave interactions with boundary layers on a flat plate were investigated experimentally in the Langley Mach 8 variable-density tunnel for plate-length Reynolds numbers ranging from 0.46×10^6 to 2.5×10^6 . The undisturbed boundary layers were laminar over the entire plate length. Pressure and heat-transfer distributions were obtained for wedge-generated incident shock waves that resulted in pressure rises ranging from 1.36 to 4.46 (both nonseparated and separated boundary-layer flows). The resulting heat-transfer amplifications ranged from 1.45 to 14. The distributions followed established trends for nonseparated flows, for incipient separation, and for laminar free-interaction pressure rises. The experimental results corroborated established trends for the extent of the pressure rise and for certain peak heat-transfer correlations. Because of the many factors that strongly affect laminar boundary layers, and because transition frequently occurred prior to the end of the interaction region, the data did not support the simpler correlations of peak heating with peak pressure that has been proposed previously.

- 87. Penland, Jim A.; and Creel, Theodore R., Jr.: Low-Speed Aerodynamic Characteristics of a Lifting-Body Hypersonic Research Aircraft Configuration. NASA TN D-7851, February 1975.**

An experimental investigation of the low-speed longitudinal, lateral, and directional stability characteristics of a lifting-body hypersonic research airplane concept was conducted in a low-speed tunnel with a 12-foot (3.66 meter) octagonal test section at the Langley Research Center. The model was tested with two sets of horizontal and vertical tip controls having different planform areas, a center vertical tail and two sets of canard controls having trapezoidal and delta planforms, and retracted and deployed engine modules and canopy. This investigation was conducted at a dynamic pressure of 239.4 Pa (5 psf) (Mach number of 0.06) and a Reynolds number of 2×10^6 based on the fuselage length. The tests were conducted through an angle-of-attack range of 0° to 30° and through horizontal-tail deflections of 10° to -30° .

88. Jones, Robert A.: Hypersonic Flight Experiments. Presented at the Hypersonic Research Engine and Hypersonic Propulsion Program, NASA Langley Research Center, March 25, 1975.
89. Penland, Jim A.; Fournier, Roger H.; and Marcum, Donald C., Jr.: Aerodynamic Characteristics of a Hypersonic Research Airplane Concept Having a 70° Swept Double-Delta Wing at Mach from 1.50 to 2.86. NASA TN D-8065, December 1975.

An experimental investigation of the static longitudinal, lateral, and directional stability characteristics of a hypersonic research airplane concept having a 70° swept double-delta wing was conducted in the Langley Unitary plan wind tunnel. The configuration variables included wing planform, tip wings, center fin, and scramjet engine modules. The investigation was conducted at Mach numbers from 1.50 to 2.86 and at a constant Reynolds number, based on fuselage, of 3.33×10^6 . Tests were conducted through an angle-of-attack range from about -4° to 24° with angles of sideslip and 0° and 3° and at elevon deflections of 0°, -10°, and -20°.

90. Combs, H. G.: Configuration Development Study of the X-24C Hypersonic Research Airplane, Phase 1. NASA CR-145032, November 1975 - March 1976.

Four hypersonic research airplane configurations found to be the most cost effective were selected for further refinement. The selection was based on a systematic analysis and evaluation of realistic designs, involving nine different configurations, evolving from three different structural/thermal concepts, coupled with existing rocket and sustainer engines. All configurations were constrained, aerodynamic envelope and maximum launch weight established by NASA.

91. Henderson, William P.: Effects of Wing Leading-Edge Radius and Reynolds Number on Longitudinal Aerodynamic Characteristics of Highly-Swept Wing-Body Configurations at Subsonic Speeds. NASA TN D-8361, December 1976.

An investigation was conducted in the Langley low-turbulence pressure tunnel to determine the effects of wing leading-edge radius and Reynolds number on the longitudinal aerodynamic characteristics of a series of highly swept wing-body configurations. The tests were conducted at Mach numbers below 0.30, angles of attack up to 16°, and Reynolds numbers per meter from 6.57×10^5 to 43.27×10^6 . The wings under study in this investigation had leading-edge sweep angles of 61.7°, 64.61°, and 67.01° in combination with trailing-edge sweep angles of 0° and 40.6°. The leading-edge radii of each wing planform could be varied from sharp to nearly round.

92. **Combs, Henry G; et al.: Configuration Development Study of the X-24C Hypersonic Research Airplane-Phase II. NASA CR-145074, January 1977.**

The X-24C Hypersonic Research Vehicle, configured with a Lockalloy heat-sink structure, a launch mass limit of 31.75 Mg and powered by an LR-105 rocket engine plus 12 LF-101 sustainer engines, has been found to be the more cost effective of the candidate configurations. In addition, the configuration provides the maximum "off design" growth potential capability and subsequently, has been selected as the candidate configuration to be subjected to the design refinement study in the remaining segment of the study. Selection of this configuration was based on the analytical study conducted on the performance growth capabilities of the candidate configurations selected from the Phase I Study.

93. **Combs, Henry G,; and et al.: Configuration Development Study of the X-24C Hypersonic Research Airplane - Phase III. NASA CR-145103, January 1977.**

The conclusion evolved from the three phased study on the configuration development of the X-24C Hypersonic Research Airplane make it evident that it is practical to design and build the high performance National Hypersonic Flight Research Facility airplane with todays state of the art within the cost and operational constraints established by NASA. The vehicle launched at 31.75 Mg from the B-52 can cruise for 40 seconds at Mach 6.78 on scramjets. Without scramjets it can approach Mach 8 with a 453.6 Kg payload or do 70 seconds of cruise at Mach 6 with a 2.27 Mg payload. Costs of the two vehicle program can be kept within \$70M. Reduction in cost is possible with a vehicle scaled to a lesser mass and capability.

94. **Bernot, Peter T.: Aerodynamic Characteristics of Two Single-Stage-To-Orbit Vehicles at Mach 20.3. NASA TM X-3550, August 1977.**

The hypersonic stability, control, and performance characteristics of two Langley-designed configurations have been determined. Each configuration (0.0025 scale) had a 50° swept delta wing, a vertical tail, and a body flap. One model represented a control-configured vehicle with a reduced level of longitudinal static stability; the other model was designed for a more conventional level of stability. Data were obtained over an angle-of-attack range of 0° to 50° and included effects of component buildup. In addition, the effects of the vertical tail on the lateral-directional characteristics were obtained over angles of attack ranging from 0° to 18°. This investigation was conducted in the 22-inch aerodynamics leg of the Langley hypersonic helium tunnel facility.

95. **Combs, Henry G.; and et al.: Configuration Development Study of the X-24C Hypersonic Research Airplane - Executive Summary. NASA CR-145274, October 1977.**

This report summarizes the results of this three phase study and makes recommendations as to the X-24C's value toward furthering hypersonic research and technology. The results show that a Lockalloy heat-sink structure affords the capability for a "work-horse" vehicle which can serve as an excellent platform for this research. It was further concluded that the performance of a blended wing body configuration surpassed that of a lifting body design for typical X-24C missions. The cost of a two vehicle program, less engines, B-52 modification and contractor support after delivery, can be kept within \$70M (in Jan. 1976 dollars).

96. **Dillon, James L.; and Pittman, James L.: Aerodynamic Characteristics at Mach Numbers From 0.33 to 1.20 of a Wing-Body Design Concept for a Hypersonic Research Airplane. NASA TP-1044, December 1977.**

An experimental investigation of the static aerodynamic characteristics of a model of one design concept for the proposed National Hypersonic Flight Research Facility was conducted in the Langley 8-foot Transonic Pressure Tunnel. The experiment consisted of configuration buildup from the basic body by adding a wing, center vertical tail, and a three-module or six-module scramjet engine. The free-stream test Mach numbers were 0.33, 0.80, 0.90, 0.95, 0.98, 1.10, and 1.20 at Reynolds numbers per meter ranging from 4.8×10^6 to 10.4×10^6 . The test angle-of-attack range was approximately -4° to 22° at constant angles of sideslip of 0° and 4° ; the angle of sideslip ranged from about -6° to 6° at constant angles of attack of 0° and 17° . The elevons were deflected 0° , -10° , and -20° with rudder deflections of 0° and 15.6° .

97. **Dillon, James L.; and Creel, Theodore R., Jr.: Aerodynamic Characteristics at Mach Number 0.2 of a Wing-Body Concept for a Hypersonic Research Airplane. NASA TP-1189, June 1978.**

An experimental investigation of the static aerodynamic characteristics of a model of a wing-body concept for a high-speed research airplane was conducted in the Langley low-turbulence pressure tunnel. The experiment consisted of configuration buildup from the basic body by adding a wing, center vertical tail, three-module scramjet, and six-module scramjet engine. The test Mach number was 0.2 at Reynolds numbers, based on fuselage length, ranging from 2.78×10^6 to 23×10^6 . The test angle-of-attack range was approximately -5° to 30° at constant angles of sideslip of 0° and 4° . The elevons were deflected from 5° to -15° . Roll and yaw control were investigated.

98. **Dillon, James L.; and Pittman, Jimmy L.: Aerodynamic Characteristics at Mach 6 of a Wing-Body Concept for a Hypersonic Research Airplane. NASA TP-1249, August 1978.**

An experimental investigation of the static aerodynamic characteristics of a 1/30-scale model of a wing-body concept for a high-speed research airplane was conducted in the Langley 20-inch Mach 6 tunnel. The investigation consisted of configuration buildup from the basic body by adding a wing, center vertical tail, three-module scramjet, and six-module scramjet engine. The test Mach number was 6 at a Reynolds number, based on model fuselage length, of about 13.7×10^6 . The test angle-of-attack range was -4° to 20° at constant angles of sideslip of 0° , -2° , and -4° . The elevons were deflected from 10° to -15° for pitch control. Roll and yaw control were investigated. Experimental aerodynamic characteristics compared with analytical estimates.

99. **Penland, James A.; Creel, Theodore R., Jr.; and Dillon, James L.: Aerodynamic Characteristics of a Hypersonic Research Airplane Concept Having a 70° Swept Double-Delta Wing at Mach Number 0.2. NASA TP-1252, September 1978.**

A wind-tunnel investigation of the static longitudinal, lateral, and directional stability characteristics of a hypersonic research airplane concept having a 70° swept double-delta wing was conducted in the Langley low-turbulence pressure tunnel. The configuration variables included wing planform, tip fins, center fin, and scramjet engine modules. The investigation was conducted at a Mach number of 0.2 over a Reynolds number (based on fuselage length) range of 2.26×10^6 to 19.75×10^6 (with a majority of tests at 10.0×10^6). Tests were conducted through an angle-of-attack range from about -2° to 34° , at angles of sideslip of 0° and 5° , and at elevon deflections of 0° , -5° , -10° , -15° , and -20° .

100. **Penland, Jim A.; Dillon, James L.; and Pittman, Jimmy L.: An Aerodynamic Analysis of Several Hypersonic Research Airplane Concepts from $M = 0.2$ to 6.0 . Journal of Aircraft, vol. 15, no. 11, November 1978.**

101. **Penland, Jim A.; Hallissy, James B.; and Dillon, James L.: Aerodynamic Characteristics of a Hypersonic Research Airplane Concept Having a 70° Swept Double-Delta Wing at Mach Numbers from 0.80 to 1.20 with Summary of Data from 0.20 to 6.0. NASA TP-1552, December 1979.**

The data of the present report are divided into three areas. The first area includes the results of a wind-tunnel investigation of the static longitudinal, lateral, and directional stability characteristics of a hypersonic research airplane concept having a 70° swept double-delta wing. The force tests were conducted in the Langley 8-foot transonic pressure tunnel for Mach numbers from 0.80 to 1.20 for a Reynolds number (based on fuselage length) range of 6.30×10^6 to 7.03×10^6 , at angles of attack from about -4° to 23° , and at angles of sideslip of 0° and 5° . The configuration variables included the wing planform, tip fins, the center vertical tail, and scramjet engine modules. The second area is a summary of the variations of the more important aerodynamic parameters with Mach number for Mach numbers from 0.20 to 6.0. The third area is a state-of-the-art example of theoretically predicting performance parameters and static longitudinal and directional stability over the Mach number range.

102. Wang, C. R.; Zakkay, V.; and Parikh, H.: An Experimental and Analytical Investigation of a Hypersonic Cruising Vehicle at Mach 6. AIAA-80-0004, January 1980.

103. Trimpi, R. L.: Modern Fluid Dynamics of Supersonic and Hypersonic Flight. AIAA Paper No. 80-0862, May 1980.

Many aspects of fluid dynamics impacting supersonic and hypersonic flight are examined. Progress, current problem areas, and prognostications for the future are discussed, with special emphasis on those fluid dynamic facets which can be more directly tied to a potential end-product vehicle. Numerous illustrative examples are included.

104. Pittman, Jimmy L.; and Riebe, Gregory D.: Experimental and Theoretical Aerodynamic Characteristics of Two Hypersonic Cruise Aircraft Concepts at Mach Numbers of 2.96, 3.96, and 4.63. NASA TP-1767, October 1980.

The longitudinal and lateral-directional aerodynamic characteristics for two Mach 5 cruise aircraft concepts have been determined for test Mach numbers of 2.96, 3.96, and 4.63. Estimates from hypersonic impact theory and first-order supersonic linearized theory were compared with data to indicate the usefulness of these methods. The method which applied tangent-cone empirical theory to the body and tangent-wedge theory to the wings and to the horizontal and vertical tails provided the best estimates. Comparisons for the configuration buildup showed, however, that the estimates for the various components could be significantly different from the data. The tangent-cone empirical theory applied to all components showed consistently poor agreement with data, and the linear theory estimates were accurate only for lift coefficient and drag coefficient at low angles of attack.

105. Small, W. J.; Riebe, G. D.; and Taylor, A. H.: Aerodynamics of a Turbojet-Boosted Launch Vehicle Concept. Journal of Spacecraft and Rockets, vol. 18, no. 1, Jan.-Feb. 1981. (Also available as AIAA Paper No. 80-0360R.)

Results from analytical and experimental studies of the aerodynamic characteristics of a turbojet-boosted launch vehicle are presented. This launch vehicle concept depends upon several novel applications of aerodynamic technology, particularly in the area of takeoff lift and minimum transonic drag requirements. The takeoff mode stresses leading edge vortex lift generated in parallel by a complex arrangement of low aspect ratio booster and orbiter wings. Wind-tunnel tests on a representative model showed that this low-speed lift is sensitive to geometric arrangements of the booster-orbiter combination and is not predictable by standard analytic techniques. Transonic drag was also experimentally observed to be very sensitive to booster location; however, these drag levels were accurately predicted by standard far-field wave drag theory.

106. **Riebe, Gregory D.; Small, William J.; and Morris, Odell A.: Aerodynamic Tests and Analysis of a Turbojet-Boosted Launch Vehicle Concept (Spacejet) Over a Mach Number Range of 1.50 to 2.86. NASA TP-1888, July 1981.**

Results from analytical and experimental studies of the aerodynamic characteristics of a turbojet-boosted launch-vehicle concept through a Mach number range of 1.50 to 2.86 are presented. The vehicle consists of a winged orbiter utilizing an area-ruled axisymmetric body and two winged turbojet boosters mounted underneath the orbiter wing. This study concentrated primarily on drag characteristics near zero lift. Force measurements and flow visualization techniques were employed. Estimates from wave drag theory, supersonic lifting surface theory, and impact theory are compared with data and indicate the ability of these theories to adequately predict the aerodynamic characteristics of the vehicle. Despite the existence of multiple wings and bodies in close proximity to each other, no large scale effects of boundary-layer separation on drag or lift could be discerned. Total drag levels were, however, sensitive to booster locations.

107. **Riebe, Gregory D.; and Pittman, Jimmy L.: Aerodynamic Characteristics of a Hypersonic Cruise Aircraft Concept with Wing-Mounted Propulsion Systems at Mach Numbers of 2.96, 3.96, and 4.63. NASA TM-81937, March 1981.**

Longitudinal aerodynamic data for a Mach 6 transport concept at Mach numbers of 2.96, 3.96, and 4.63 are presented in this report. The model components consisted of a lenticular-cross-section fuselage, a wedge-slab-wedge wing, a horizontal-tail set, a wedge center vertical tail, a set of flow-through ramjet nacelles mounted under the wing, and a set of turbojet nacelles simulated by solid bodies mounted on the upper surface of the wing. Estimates from hypersonic impact theory were compared with experimental data for several of the test configurations. The method, which applied tangent-cone empirical theory to the body and tangent-wedge theory to the planar surfaces, generally provided a good estimate of the fuselage-wing-tail data. Comparisons, however, for the configuration buildup showed that the theoretical estimates for the various components could be significantly different from the experimental data, which is an indication of errors in the local pressure estimates.

108. **Wilhite, A. W.: Optimum Wing Sizing of a Single-Stage-to-Orbit Vehicle. AIAA Paper No. 82-0174, January 1982.**

An investigation has been conducted to determine preliminary wing designs for a single-stage-to-orbit (SSTO) vehicle. This vehicle has the following mission profile: vertical takeoff, boost-to-orbit, hypersonic reentry, and horizontal landing. For this vehicle, the wing is sized to meet Space Shuttle reentry aerodynamic requirements for hypersonic trim and horizontal landing, since reentry trajectories for the Shuttle and the SSTO vehicle are similar. A hypersonic and subsonic aerodynamic computer program was developed and combined with an existing optimization algorithm to automatically size and shape a wing which satisfies both reentry and landing requirements while also maintaining a minimum mass design. With this procedure, the influence of hypersonic and subsonic aerodynamic requirements, control surface size, and center-of-gravity positions on the initial wing design were investigated.

- 109. Bernot, Peter T.: Effect of Modifications on Aerodynamic Characteristics of a Single-Stage-to-Orbit Vehicle at Mach 5.9. NASA TM-84565, January 1983.**

Hypersonic stability, control, and performance characteristics were determined on a single-stage-to-orbit vehicle based on control-configured stability concepts. The configuration (0.006-scale model) had a large body with a small 50 deg swept wing. Two vertical-fin arrangements were investigated which consisted of a large centerline vertical tail and small wing-tip fins. The wing-tip fins had movable surfaces called controllers which could be deflected outward. Longitudinal and lateral directional characteristics were obtained over an angle-of-attack range from 0 deg to 40 deg. The effects of tip-fin controller deflection on roll- and yaw-control characteristics at a sideslip of angle of 0 deg were obtained. This investigation was conducted in the Langley 20 inch Mach 6 tunnel.

- 110. Wurster, K. E.; and Zoby, E. V.: An Experimental Heat-Transfer Investigation of an Advanced Winged Entry Vehicle at Mach 10. AIAA Paper 83-0409, January 1983.**

Heat-transfer measurements have been made at Mach 10 in air on an instrumented 0.006-scale model of an advanced winged entry vehicle. Data were obtained at 83 thermocouple stations which include locations on the lower and upper surface centerlines, spanwise positions along the lower and upper surfaces on the wing, the lower surface of the body flap, and radial locations on the fuselage. Data were obtained for angles of attack ranging from 0 to 45 deg, sideslip angles of + or - 2 deg. Reynolds numbers of 0.5, 1.0, and 2.0 million per foot, and body-flap deflections of 1, 10, and 20 deg. The data generally indicate increased windward heating and decreased leeside heating with increased angle of attack, significantly increased body-flap heating with deflection angle, and minor variations in heating with sideslip, increasing in magnitude with angle of attack. Windward centerline data are shown to be in fair agreement with results of predictions based on an approximate engineering method.

- 111. Beach, H. Lee, Jr.; Mackley, Ernest A.; and Guy, Robert W.: Mach 7 Tests of Langley Airframe-Integrated Scramjet. NASA TM-84595, June 1983.**

Engine test results at simulated Mach 7 conditions are presented for an airframe-integrated supersonic combustion ramjet concept. Tests were conducted at the NASA Langley Research Center and at the General Applied Science Laboratories, Inc. over a simulated flight dynamic pressure range of 19 to 48 kN/m². Combustion problems that arose during early tests of the subscale (20-cm-high) model were addressed by geometric changes that locally raised temperatures and pressures to enhance ignition and flameholding. Subsequent tests utilized monosilane (SiH₄) as an ignitor and staged injection for flameholding. Scramjet engine internal performance results are presented.

- 112. Penland, Jim A.; Marcum, Don C., Jr.; and Stack, Sharon H.: Wall-Temperature Effects on the Aerodynamics of a Hydrogen-Fueled Transport Concept in Mach 8 Blowdown and Shock Tunnels. NASA TP-2159, July 1983.**

Results are presented from two separate tests on the same blended wing-body hydrogen-fueled transport model at a Mach number of about 8 and a range of Reynolds numbers (based on theoretical body length) of 0.597×10^6 to about 156.22×10^6 . Tests were made in a conventional hypersonic blowdown tunnel and a hypersonic shock tunnel at angles of attack of -2° to about 8° , with an extensive study made at a constant angle of attack of 3° . The model boundary-layer flow varied from laminar at the lower Reynolds numbers to predominantly turbulent at the higher Reynolds numbers. Model wall temperatures and stream static temperatures varied widely between the two tests, particularly at the lower Reynolds numbers. These temperature differences resulted in marked variations of the axial-force coefficients between the two tests, due in part to the effects of induced pressure and viscous interaction variations. The normal-force coefficient was essentially independent of Reynolds number. Analysis of results utilized current theoretical computer programs and basic boundary-layer theory.

- 113. Riebe, Gregory D.: Aerodynamic Characteristics, Including Effect of Body Shape of a Mach 6 Aircraft Concept. NASA TP-2235, December 1983.**

Longitudinal aerodynamic characteristics for a hydrogen-fueled hypersonic transport concept at Mach 6 are presented in this report. The model components consist of four bodies with identical longitudinal area distributions but different cross-sectional shapes and widths, a wing, horizontal and vertical tails, and a set of wing-mounted nacelles simulated by solid bodies on the wing upper surface. Lift-drag ratios were found to be only slightly affected by fuselage planform width or cross-sectional shape. Relative distribution of fuselage volume above and below the wing was found to have an effect on the lift-drag ratio, with a higher lift-drag ratio produced by the higher wing position.

- 114. Wurster, K. E.; and Zoby, E. V.: An Experimental Investigation of Surface Pressure Measurements on an Advanced Winged Entry Vehicle at Mach 10. AIAA Paper 84-0308, January 1984.**

Surface pressure measurements have been made at Mach 10 in air on an instrumented 0.006-scale model of an advanced (control configured) winged entry vehicle. The tests were conducted in the Langley continuous flow Hypersonic Tunnel. Data were obtained at 83 surface pressure stations, which include locations on the lower and upper surface centerlines, spanwise positions along the lower and upper surfaces of the wing, the lower surface of the body flap, and radial locations on the fuselage. Data were obtained for angles of attack ranging from zero to 40° , sideslip angles of -2° to $+5^\circ$. Reynolds numbers of 0.5, 1.0, and 2.0 million per foot, and body-flap deflections of zero, 10° , and 20° . Test conditions and orifice locations were chosen to correspond directly with those for the heat transfer measurements.

- 115. Wilhite, A. W.; Arrington, J. P.; and McCandless, R. S.: Performance Aerodynamics of Aero-Assisted Orbital Transfer Vehicles. AIAA Paper 84-0406, January 1984.**

116. **Bernot, Peter T.: Exploratory Investigation of the Aerodynamic Characteristics of a Biwing Vehicle at Mach 20.3. NASA TM-86314, February 1984.**

Longitudinal and lateral-directional characteristics of a simple biwing configuration were determined over an angle-of-attack range from -3 deg to 50 deg. The body was comprised of a cylindrical section with an ogival forebody having an overall fineness ratio of 6.67. The delta wings had a 38.3 deg sweep angle and were geometrically similar in planform. The upper wing was located slightly forward relative to the lower wing. The model was tested in upright and inverted orientations including component buildups. This investigation was conducted in the 22-inch aerodynamics leg of the Langley Hypersonic Helium Tunnel Facility.

117. **Powell, R. W.; Talay, T. A.; Wilhite, A. W.; Rehder, J. J.; White, N. H.; Naftel, J. C.; Stone, H. W.; Arrington, J. P.; and McCandless, R. S.: Performance Assessment of Aero-Assisted Orbital Transfer Vehicles. NASA CP-2347, OTV Propulsion Issues Conference, April 1984.**
118. **Scallion, William I.; and Phillips, W. Pelham: Space Shuttle Orbiter Trimmed Center-of-Gravity Extension Study. Summary Report. NASA TP-2284, 1984.**
119. **Lamb, M.: Experimental and Theoretical Aerodynamic Characteristics of Two Mach 4 Cruise Aircraft Concepts at Mach Numbers From 1.50 to 4.63 . NASA TM 85801, July 1984. CONFIDENTIAL**
120. **Helms, Vernon T., III: Oil-Flow Study of a Space Shuttle Orbiter Tip-Fin Controller. NASA TM-86276, December 1984.**
121. **McCandless, R. S.; and Cruz, C. I.: Hypersonic Characteristics of an Advanced Aerospace Plane. AIAA Paper 85-0346, January 1985.**

A series of hypersonic wind-tunnel tests have been conducted in the NASA Langley Hypersonic Facilities Complex to obtain the static longitudinal and lateral-directional aerodynamic characteristics of an advanced aerospace plane. Data were obtained at 0 to 20 deg angles of attack and -3 to 3 deg angles of sideslip at Mach numbers of 6 and 10 in air and 20 in helium. Results show that stable trim capability exists at angles of attack near maximum lift-drag ratio (L/D). Both performance and stability exhibited some Mach number dependency. The vehicle was longitudinally unstable at low angles of attack but stable at angles of attack near and above maximum L/D. It was directionally unstable with positive dihedral effect. The rudder showed an inability to provide lateral-directional control, and removing the vertical tail resulted in increased directional instability. Analytical predictions of the static longitudinal aerodynamic coefficients gave relatively good comparisons with the experimental data.

122. **Powell, R. W.; and Scallion, W. I.: Effects of Canards on the Landing Characteristics of the Space Shuttle Orbiter. NASA TM-86305, January 1985. CONFIDENTIAL**

A preliminary feasibility study has been performed to evaluate the impact of using canards on the Space Shuttle orbiter during the landing phase. The study showed that the addition of the canards significantly reduces touchdown speed when compared with the current orbiter by allowing the elevons to be set at a more positive deflection angle, thereby increasing lift at the same angle of attack.

123. **Wells, William L.; MacConochie, Ian O.; Helms, Vernon T., III; and Raney, David: Heating Rate Distributions at Mach 10 on a Circular Body Earth-To-Orbit Transport Vehicle. AIAA Paper 85-0974, June 1985.**

Among the concepts being considered for future Earth-to-orbit transport vehicles are fully reusable single-stage systems which take off vertically and land horizontally. Because these vehicles carry their own propellant internally, they are much larger than the present Space Shuttle Orbiter. One such single-stage vehicle under study is the circular body configuration which has the advantages of simple structural design and large volume-to-weight ratio. As part of an overall evaluation of this configuration, a series of heat transfer and surface flow tests were conducted. The phase-change paint and oil-flow tests were performed in the Langley 31-Inch Mach-10 Tunnel at angles of attack from 20 through 40 degrees in 5-degree increments. Heat-transfer coefficient data are presented for all angles of attack and detailed oil-flow photographs are shown for windward and leeward surfaces at 25 and 40 degrees angle of attack. In many ways, heating was similar to that previously determined for the Shuttle Orbiter so that, in a cursory sense, existing thermal protection systems would appear to be adequate for the proposed circular-body configurations.

124. **Penland, Jim A.; and Pittman, James L.: Aerodynamic Characteristics of a Distinct Wing-Body Configuration at Mach 6 - Experiment, Theory, and the Hypersonic Isolation Principle. NASA TP 2467, August 1985.**

An experimental investigation has been conducted to determine the effect of wing leading-edge sweep and wing translation on the aerodynamic characteristics of a wing-body configuration at a free-stream Mach number of about 6 and Reynolds number (based on body length) of 17.9×10^6 . Seven wings with leading-edge sweep angles from -30° to 60° were tested on a common body over an angle-of-attack range from -12° to 10° . All wings had a common span, aspect ratio, taper ratio, planform area, and thickness ratio. Wings were translated longitudinally on the body to make tests possible with the total and exposed mean aerodynamic chords located at a fixed body station. Aerodynamic forces were found to be independent of wing sweep and translation, and pitching moments were constant when the exposed wing mean aerodynamic chord was located at a fixed body station. Thus, the "Hypersonic Isolation Principle" was verified. Theory applied with tangent-wedge pressures on the wing and tangent-cone pressures on the body provided excellent predictions of aerodynamic force coefficients but poor estimates of moment coefficients.

125. Micol, J. R.; and Wilmuenster, K. J.: **Experimental Aerodynamic Coefficient on a Shuttle-Like Vehicle at Mach 6 and 10 and Comparison to Prediction.** AIAA Paper 85-1796, August 1985.
126. Young, J. C.; and Findlay, J. T.: **An Analysis of the Space Shuttle Hypersonic Entry Trim Prediction Anomaly.** AIAA Paper 85-1764-CP, August 1985.

This paper reviews a parameter identification methodology developed to investigate the hypersonic longitudinal trim misprediction apparent in the NASA Space Shuttle Orbiter entry flights. The method combines an analysis using a measured versus predicted technique in conjunction with a multilinear regression analysis to identify prediction deficiencies using quasi-static longitudinal data in the hypersonic flight regime (Mach 6 through 26). In general, the results of this extraction confirm results previously obtained by other Shuttle investigators with the exception of elevon results. A combination of this analytical tool and other flight data will enable flight data interpretation with the potential for identifying the sources of the Shuttle's hypersonic trim misprediction to an accuracy consistent with updating preflight prediction methodology for future spacecraft.

127. Johnson, S. K.; and Weidner, J. P.: **Experimental and Theoretical Investigation of a Parametric Set of Cruise Configurations at High Supersonic Speeds.** (U) NASA TM 86357, 1985. CONFIDENTIAL
128. McCandless, R. S.: **Hypersonic Characteristics of an Advanced Aerospace Plane at Mach 20.3.** NASA TM-86435, December 1985.

Wind-tunnel studies have been performed in the Langley Hypersonic Helium Tunnel Facility to obtain static longitudinal and lateral-directional aerodynamic characteristics of an advanced aerospace plane concept. The nominal test conditions are a Mach number of 20.3 and a Reynolds number of 6.8×10 to the 6th power per foot at angles of attack from 0 to 25 deg and angles of sideslip of -3 and 0 deg. Stability and control characteristics are obtained for several deflections of the elevators, elevons, and rudder. In addition, a modified canopy is examined. The results indicate that this vehicle is longitudinally stable at angles of attack near the maximum lift-drag ratio. Also, the vehicle is shown to be directionally unstable with positive dihedral effect.

129. Brauckmann, Gregory J.: **Hypersonic Characteristics of a Maneuvering Reentry Research Vehicle.** AIAA Paper 86-0271, January 1986.

Static longitudinal and lateral-directional aerodynamic characteristics of a candidate lifting entry research vehicle were measured at Mach numbers of 6 and 10 in air and at Mach 20 in helium. The effects of nose bluntness, body flap deflection, and elevon deflection were examined over a range of angle of attack. The vehicle demonstrated static longitudinal and lateral-directional stability at angles of attack from the maximum lift-to-drag ratio $(L/D)_{\max}$ to high angles of attack. Sufficient control effectiveness existed to provide longitudinal trim from $(L/D)_{\max}$ to high angles of attack. Aerodynamic coefficients predicted using the Hypersonic Arbitrary Body Program were compared to measurement. Good agreement was obtained for lift-to-drag values.

130. Johnson, S. K.; and Weidner, J. P.: Aerodynamic Characteristics of a Parametric Set of Mach 5 Cruise Configurations with Spilling Nacelles at Supersonic Speeds. NASA TP 2516, March 1986. CONFIDENTIAL
131. Brauckmann, Gregory J.; and Woods, William C.: Aerodynamic Characteristics of a Maneuvering Reentry Research Vehicle at Mach 20.3. NASA TM 87643, April 1986.
132. Brauckmann, Gregory J.; and Blackstock, Thomas A.: Aerodynamic Characteristics of a Maneuvering Reentry Research Vehicle at Mach 10.2. NASA TM-87657, April 1986.
133. Wilmuenster, K. J.; and Hamilton, H. H.: A Comparison of Computed and Measured Aerodynamic Characteristics of a Proposed Aeroassist Flight Experimental Configuration. Presented at the AIAA/ASME 4th Joint Thermophysics and Heat Transfer Conference, Boston, Massachusetts, June 2-4, 1986.

SUBJECT CATEGORY 3 - AERO-HEATING

1. **McLellan, C. H.: An Investigation of the Melting of Bodies Due to Aerodynamic Heating. Transactions of the Am. Soc. of Mechanical Engineers, vol. 77, July 1955, p. 727-734.**

Various approaches to the aerodynamic heating problem of bodies reentering the atmosphere are considered briefly. One approach to this problem is to allow the heat to be absorbed by the body. At very high supersonic velocities, part of the body can be expected to melt. An analysis of a high-density conical-nosed vehicle entering the atmosphere at a speed of 20,200 fps showed that, in addition to the melting at the tip of the cone, the temperature gradients through the skin become very large and, under some conditions, surface melting would occur on the cone as well. Since very little was known about the melting of bodies due to aerodynamic heating, an exploratory investigation was made in the Langley 11-inch hypersonic tunnel. The results of melting tests of several bodies made of a low-melting-temperature alloy are discussed.

2. **Becker, John V.; and Korycinski, Peter F.: Heat Transfer and Pressure Distribution at a Mach Number of 6.8 on Bodies With Conical Flares and Extensive Flow Separation. NACA RM L56F22, August 1956.**

An investigation of heat transfer and pressure distribution on flare bodies under conditions of laminar, transitional, and turbulent boundary layers was conducted in the Langley 11-inch hypersonic tunnel at a Mach number of 6.8. Analysis of the results deals with factors affecting the extent of separation, heat transfer in the separated zone and after reattachment, and the adequacy of pertinent theoretical methods.

3. **Crawford, Davis H.; and McCauley, William D.: Investigation of the Laminar Aerodynamic Heat-Transfer Characteristics of a Hemisphere-Cylinder in the Langley 11-Inch Hypersonic Tunnel at a Mach Number of 6.8. NACA Report 1323, 1957. (Supersedes NACS TN 3706, March 1956).**

The aerodynamic heat transfer of an isothermal hemisphere-cylinder has been investigated at a Mach number of 6.8 and Reynolds numbers from 0.14×10^6 to 1.06×10^6 based on diameter and free-stream conditions. The experimental heat transfer was slightly less than that predicted by the Stine and Wanlass theory (NACA TN 3344) for an isothermal surface. For stations within 45° of the stagnation point the heat transfer could be correlated by a single relation between local Stanton and local Reynolds numbers.

4. **Crawford, Davis H.; and Rumsey, Charles B.: Heat Transfer in Regions of Separated and Reattached Flows. NACA RM L57D25B, June 1957.**

This paper considers the effects of separation on heat transfer and has indicated a method for reducing separation. The heat transfer in regions of separation and reattachment for a few specific shapes are given. These results show that the heat transfer in a separated region is strongly affected by the extent of separation, the location of the reattachment point, and the location of transition along the separated boundary.

5. **Feller, William V.: Heat Transfer to Bodies at Angles of Attack. NACA RM L57D19c, June 1957.**

Heat-transfer rates were measured on a modified Karman nose shapes at an angle of attack of 0° at Mach numbers of 6.86 and 3.69, and at angles of attack up to 25° at a Mach number of 3.69. Data are presented for a smooth model, which showed natural transition, and for the model with roughness strips, which caused fully turbulent flow.

6. **Bond, A. C.; Lee, D. B.; and Rumsey, C. B.: Skin Temperature and Surface Pressure Of Blunted Cone-Cylinder-Flare Configuration Free Flight Test Vehicle To Hypersonic Speeds. NACA RM-L58G21, September 1958.**

7. **Becker, J. V.; and Korycinski, P. F.: The Surface Coolant Requirements of Hypersonic Gliders. NASA MEMO, April 1959.**

8. **Bertram, Mitchel H.; and Feller, William V.: A Simple Method for Determining Heat Transfer, Skin Friction, and Boundary-Layer Thickness for Hypersonic Laminar Boundary-Layer Flows in a Pressure Gradient. NASA Memo 5-24-59L, June 1959.**

A procedure based on the method of similar solutions is presented by which the boundary-layer effects in a laminar hypersonic flow may be rapidly evaluated if the pressure distribution is known. This solution, which at present is restricted to power-law variations of pressure with surface distance, is presented for a wide range of exponents in the power law corresponding to both favorable and adverse pressure gradients. This theory has been compared with results from heat-transfer experiments on blunt-nose models at free-stream Mach numbers of 4 and 6.8, including tests at angle of attack. Correlation of boundary-layer-thickness data has also been shown.

9. **Crawford, Davis H.: Investigation of the Flow Over a Spiked-Nose Hemisphere-Cylinder at a Mach Number of 6.8. NASA TN D-118, December 1959.**

The shape and nature of the flow over a spiked-cone hemisphere-cylinder were studied in detail at a nominal Mach number of 6.8 and in a Reynolds number range (based on diameter and stream conditions head of the model) of 0.12×10^6 to 1.5×10^6 . Schlieren photographs showed the effect of varying the spike length and Reynolds number upon the shape of the separated boundary and upon the location of transition. The heat transfer and pressure distribution over the body were then correlated with the location of the start of separation, the location of reattachment, and the location of the start of transition.

10. **Stainback, P. C.: Preliminary Heat-Transfer Measurements On A Hypersonic Glide Configuration Having 79.5 deg. Sweepback and 45 deg. Dihedral At A Mach Number Of 4.95. NASA TM X-247, February 1960.**

11. **Reed, Robert D.; and Watts, Joe D.: Skin and Structural Temperatures Measured on the X-15 Airplane During a Flight to a Mach Number of 3.3. NASA TM X-468, January 1961.**

Temperatures obtained from thermocouple data are presented for skin and structural areas of the fuselage, wing, horizontal tail, and vertical tail. No heat-transfer analysis is included. The maximum skin temperature recorded was 440°F. A temperature-indicating paint was used to illustrate visually the temperature variations over the lower surface of the wing.

12. **Stainback, P. C.: Heat-Transfer Measurements At A Mach Number Of 4.95 On Two 60 deg. Swept Delta Wings With Blunt Leading Edges and Dihedral Angles Of 0 deg. and 45 deg. NASA TN D-549, January 1961.**

13. **Beckwith, Ivan E.; and Gallagher, James J.: Local Heat Transfer and Recovery Temperatures on a Yawed Cylinder at a Mach Number of 4.15 and High Reynolds Numbers. NASA TR R-104, 1961. (Supersedes NASA MEMO 2-27-59L.)**

14. **Newlander, Robert A.: Effect of Shock Impingement on the Distribution of Heat-Transfer Coefficients on a Right Circular Cylinder at Mach Numbers of 2.65, 3.51, and 4.44. NASA TN D-642, January 1961.**

A 2.8-inch-diameter cylinder was placed downstream of a 16-1/4° wedge on a flat-plate test surface which was mounted so as to produce boundary-layer thicknesses of 6 and 0.6 inches. The maximum heat-transfer coefficient on the stagnation line of the cylinder increased 1-1/2 to 3 times for laminar theoretical value for a cylinder of infinite length at Mach numbers of 2.65 and 4.44, respectively. The maximum heating rate on the adjacent flat-plate surface immediately upstream of the cylinder increased from approximately 4 times the undisturbed flat-plate heat transfer at a Mach number of 2.65 to approximately 10 times the undisturbed flat-plate value at a Mach number of 4.44.

15. **Rumsey, Charles B.; and Lee, Dorothy B.: Measurements of Aerodynamic Heat Transfer at Boundary-Layer Transition on a 10° Cone in Free Flight at Supersonic Mach Numbers Up to 5.9. NASA TN D-745, May 1961.**

The range of local Reynolds number was from 6×10^6 to 55.2×10^6 . In general, measurements were in good agreement with theory for cones. At a relatively constant ratio of wall temperature to local static temperature near 1.2, the transition Reynolds number increased from 9.2×10^6 to 19.4×10^6 as Mach number increased from 1.57 to 3.38. At Mach numbers near 3.7, the transition Reynolds number decreased as the skin temperature increased toward adiabatic wall temperature.

16. **Bland, W. M., Jr.; and Collie, K. A.: Free-Flight Aerodynamic-Heating Data to Mach Number 10.4 For A Modified Von Karman Nose Shape. (Supersedes NACA RM L56025), NASA TN D-889, May 1961.**

17. **Carter, Howard S.; and Carr, Robert E.: Free-Flight Investigation of Heat Transfer to an Unswept Cylinder Subjected to an Incident Shock and Flow Interference From an Upstream Body at Mach Numbers Up to 5.50. NASA TN D-988, 1961.**

18. **Dunavant, J. C.: Investigation of Heat Transfer and Pressures on Highly Swept Flat and Dihedral Delta Wings at $M = 6.8$ and 9.6 at Angles of Attack to 90° . NASA TM X-688, Sept. 1961.**
19. **Becker, J. V.; and Korycinski, P. F.: Heat Transfer and Pressure Distribution on Bodies With Conical Flares and Extensive Flow Separation. NASA-TN-D-1260.**
20. **Jones, Robert A.; and Trimpi, Robert L.: Heat-Transfer and Pressure Distributions at a Mach Number of 6 for 70° Swept Slab Wings With Sharp and Spherical Noses and Cylindrical Leading Edges. NASA TM X-682, May 1962.**

Local heat-transfer coefficients and surface pressures in the region of the cylindrical leading edge of 70° swept delta wings with sharp and spherical noses were determined from wind-tunnel tests at a Mach number of 6; free-stream Reynolds numbers, based on leading-edge diameter of 3×10^4 , 6×10^4 , and 23×10^4 ; and angles of attack from 0° to 30° . Within the range of 3 to 6 leading-edge diameters from the nose at angles of attack to 20° , the heat transfer to the leading edge of the sharp-nose wing was found to agree with values predicted by an easily applied theory. At angles of attack of 25° and 30° , the high Reynolds number data for the windward side of the leading edge indicate transitional or turbulent flow with measured heating rates much higher than theoretical values. Within the range of 2 to 5 leading-edge diameters from the nose, the heat transfer to the leading edge of the spherical-nose wing was found to be slightly higher than for the sharp-nose-wing with higher heating rates for locations nearer the nose. In addition, the heat-transfer-coefficient ratios were found to be higher for the low Reynolds number data for locations near the nose than for the high Reynolds number data.

21. **Bertram, M. H.; and Wiggs, M. Margarete: Effect of Surface Distortions on the Heat Transfer to a Wing at Hypersonic Speeds. Inst. Aerospace Sci. Paper 62-127, June 1962.**

Sources of disturbances to the surface flow on hypersonic vehicles may result from local distortions due to aerodynamic heating effects on the structure and skin. The effects of surface distortions, consisting of slots, small steps, and local curvature, on the heat transfer and surface pressures on a wing with a thick boundary layer at hypersonic speeds are examined. The results of experiments on an unswept slab wing at nominal free-stream Mach numbers of 7 and 10 show the effect of these surface distortions on heating and surface pressure at angles of attack up to 20° .

22. **Caldwell, A. L.; Hougseth, E. G.; and Miller, D. S.: The Influence of Aerodynamic Interference Heating on Directional Stability Problems of Hypersonic Vehicles. Paper 63-3, Inst. Aerospace Sci., January 1963.**

23. **Peterson, John B., Jr.: A Comparison of Experimental and Theoretical Results for the Compressible Turbulent-Boundary-Layer Skin Friction With Zero Pressure Gradients. NASA TN D-1795, March 1963.**

Seven theories of compressible turbulent-boundary-layer skin friction was compared with published experimental results in order to determine their accuracy. Comparison with published data taken at Mach numbers of 1.5 to 10 and Reynolds numbers of 2 to 100×10^6 indicates that the Sommer and Short T' method most accurately matches experimental data. Curves to aid in the calculation of compressible turbulent skin friction by the Sommer and Short T' method are presented.

24. **Jones, Robert A.: Heat-Transfer and Pressure Distributions of Two Flat-Face Sharp-Cone Bodies of Revolution at a Mach Number of 8 and Comparison With Data for a Round Corner Body. NASA TM X-774, May 1963.**

Heat-transfer and pressure distributions of two flat-face sharp-corner bodies of revolution obtained at a Mach number of 8 for angles of attack from 0° to 45° are presented and compared with data of a similar shape having a rounded corner with a ratio of corner radius to face radius of 0.2. The results indicate that a sharp corner causes a more rapid movement of the stagnation-point location with angle of attack and also tends to promote local separation at the corner. The value of the heat-transfer coefficient of the stagnation point at zero angle of attack agreed with that predicted by theory and, although the maximum heating rate would be higher for a sharp-corner configuration, the total heat load to the face would be less than that for a rounded-corner configuration.

25. **Watts, J. D.; and Banas, R. P.: X-15 Structural Temperature Measurements and Calculations for Flights to Maximum Mach Numbers of Approximately 4, 5, and 6. NASA TM X-883, August 1963.**

26. **Sterrett, James R.; and Holloway, Paul F.: Effects of Controlled Roughness on Boundary-Layer Transition at a Mach Number of 6.0. AIAA Journal, vol. 1, no. 8, August 1963.**

An investigation of hypersonic boundary-layer transition on a smooth flat plate with a sharp leading edge (leading edge thickness less than 0.022 in.) and with or without controlled roughness is discussed. The results of the investigation are summarized in tabular form, and the experimental "critical" roughness Reynolds numbers are compared with those found in previous studies at lower supersonic Mach numbers. The critical roughness Reynolds numbers increased sharply for the higher Mach numbers studied (4.8 to 6.0) and show experimentally that the roughness parameter required to move the beginning of fully developed turbulent flow approximately to the roughness position is much greater for high supersonic and hypersonic Mach numbers than that previously established for low supersonic Mach numbers.

27. **Dunavant, J. C.: Heat Transfer To A Delta-Wing-Half-Cone Combination At Mach Numbers Of 7 and 10. NASA TN D-2199, March 1964.**

28. **Holloway, Paul F.; and Sterrett, James R.: Effect of Controlled Surface Roughness on Boundary-Layer Transition and Heat Transfer at Mach Numbers of 4.8 and 6.0. NASA TN D-2054, April 1964.**

A systematic investigation has been conducted to determine the effects of various size spheres alined in a row equidistant from the leading edge on boundary-layer transition and the critical roughness Reynolds number increase with increasing Mach number (above a Mach number of approximately 3.5 to 4.0). Calculations of the heat-transfer distributions based on simple flat-plate theory are shown to give a reasonably good prediction of the experimental results.

29. **Dunavant, J. C.: Heat Transfer to a Delta Wing—Half-Cone Combination at Mach Numbers of 7 and 10. NASA TN D-2199, March 1964.**

The results of the investigation of this configuration showed no effect of any shock from the cone on the wing surface pressure or heat transfer. The heat transfer to the stagnation line of the cone is little affected by the presence of the wing. The heat transfer to the wing surface showed agreement with the appropriate laminar or turbulent theory using measured pressures and based on a strip type of flow from the leading edge. Transition, which may be the result of the vortex near the corner, was observed at Reynolds numbers of less than 0.5×10^6 .

30. **Jones, Robert A.: Heat--Transfer and Pressure Investigation of a Fin-Plate Interference Models at a Mach Number of 6. NASA TN D-2028, July 1964.**

A 60° swept cylindrical leading-edge fin mounted on a sharp flat plate was investigated at a Mach number of 6 over a range of Reynolds numbers, based on free-stream conditions and fin leading-edge diameter, from 0.062×10^6 to 0.77×10^6 . The plate was maintained at zero angle of attack and the yaw angle of the final was varied from 0° to 30° . A relatively weak shock wave which originated at the leading edge of the plate impinged on the leading edge of the fin. Heat-transfer rates and pressures were measured on both the plate and the fin. The measured data on the fin and plate are compared with values calculated from laminar and turbulent theories for an infinitely long 60° swept cylinder and undisturbed plate.

31. **Jones, Robert A.; and Hunt, James L.: Use of Temperature Sensitive Coatings for Obtaining Quantitative Aerodynamic Heat-Transfer Data. AIAA Journal, vol. 2, no. 7, August 1964, pp. 1354-1356.**

Since the time that a temperature-sensitive coating was first used at the Langley Research Center for determining qualitative aerodynamic heating rates (early 1959), a development program has been underway to perfect a technique whereby quantitative data could be obtained by this method. This article presents some of the findings from this program.

32. **Hunt, Larry R.; and Howell, Roane R.: Experimental Technique for Measuring Total Aerodynamic Heating Rates to Bodies of Arbitrary Shape with Results for Mach 7. NASA TN D-2446, September 1964.**

33. **Dunavant, J. C.; and Everhart, P. E.: Heat Transfer Distribution on 70 deg Swept Slab Delta Wings At A Mach Number of 9.86 and Angles of Attack Up To 90 deg. NASA TN D-2302, October 1964.**
34. **Jones, R. A.; and Hunt, J. L.: An Improved Technique for Obtaining Quantitative Aerodynamic Heat-Transfer Data with Surface Coating Materials. AIAA Aerospace Sciences Meeting, New York, January 1965.**
35. **Jones, Robert A.; and Hunt, James L.: Effects of Cavities, Protuberances, and Reaction Control Jets on Heat Transfer to the Apollo Command Module. NASA TM X-1063, March 1965.**
36. **Whitehead, Allen H. and Dunavant, James C.: A Study of Pressure and Heat Transfer Over an 80° Sweep Slap Delta Wing in Hypersonic Flow. NASA TN D-2708, March 1965.**

The wing has cylindrical leading edges, a hemispherical nose of the same radius, and a length of 7.12 nose diameters. The angle of attack varied from 0° to 35° . Reynolds numbers of 9.9×10^4 and 37.0×10^4 based on model thickness were obtained at Mach 6.8, and a Reynolds number of 10×10^4 was employed for the Mach 9.6 tests. The nose greatly influenced the flow behavior over the leading-edge region at low angles of attack.

37. **Jones, R. A.; and Hunt, J. L.: Recent Experimental Studies on Heat-Transfer to the Apollo Command Module. Conference on Langley Research Related to Apollo, NASA SP-101, June 1965, pp. 9-18.**
38. **Adcock, Jerry B.; Peterson, John B., Jr.; and McRee, Donald I.: Experimental Investigation of a Turbulent Boundary Layer at Mach 6, High Reynolds Numbers, and Zero Heat Transfer. NASA TN D-2907, July 1965.**

An experimental investigation of the turbulent boundary layer on a hollow cylinder was made at a Mach number of 6 and with adiabatic wall temperature at Reynolds numbers based on the distance from the leading edge from 5×10^6 to 33×10^6 . Data obtained include turbulent recovery factors, a total temperature profile, velocity profiles, and average skin-friction coefficients by the momentum method. The turbulent recovery factor was found to be approximately 0.88; a slight decrease of recovery factor with increased in Reynolds number was obtained.

39. **Holloway, Paul F.; Sterrett, James R.; and Creekmore, Helen S.: An Investigation of Heat Transfer Within Regions of Separated Flow at a Mach Number of 6.0. NASA TN D-3074, November 1965.**

An extensive systematic investigation of the heat transfer associated with regions of laminar, transitional, and turbulent separation has been conducted on sharp- and blunt-leading-edge flat plates at a Mach number of 6.0 over a free-stream unit Reynolds number range of approximately 1×10^6 to 8×10^6 per foot. Separated regions were forced by forward- and rearward-facing steps, and by 10° , 20° , 30° , and 40° wedges located in several longitudinal positions on the plate.

40. **Bushnell, Dennis M.: Interference Heating on a Swept Cylinder in Region of Intersection with a Wedge at Mach 8. NASA TN D-3094, 1965.**

Local heat-transfer rates and pressures have been measured on a cylinder in the interference flow region between the cylinder and a 12° half-angle wedge. The tests were conducted at a Mach number of 8 with the cylinder at sweep angles of 45° and 60° with respect to the free stream. Results indicate that for both sweep angles, local heating is increased along the portion of the cylinder subjected to the wedge flow, but the maximum increase can be predicted for both laminar and turbulent boundary-layer flow by using local wedge flow conditions in the infinite swept-cylinder theories. Because of the higher local Reynolds number in the wedge flow region, transition occurs along the cylinder stagnation line at a lower value of free-stream Reynolds number for the cylinder-wedge configuration than for the undisturbed cylinder; however, the local Reynolds number for transition based on conditions ahead of the cylinder bow shock and the cylinder diameter was virtually the same for all configurations tested.

41. **Bushnell, Dennis M.; and Jones, Robert A.: Heat Transfer to a 70° Swept Fin Partially Submerged in a Turbulent Boundary Layer at Mach 6. NASA TM X-1191, 1965.**

42. **Fitzsimmons, H. D.; and Doyle, L. B.: Analysis of Hypersonic Pressure and Heat Transfer Tests on Delta Wings With Laminar and Turbulent Boundary Layers. CR-153994, 1965.**

Results are presented of an analysis of slab delta wing pressure and heat transfer data with laminar and turbulent boundary layers. The data were obtained during the X-20 (Dyna-Soar) program from a parametric series of model tested in conventional wind tunnels at Mach numbers of 6, 7, and 8. Shock tunnel data at Mach numbers of 6 and 15 and shock tube data at a Mach Number of 2.2 are also presented. All tests were in air. Free stream Reynolds numbers based on leading edge diameter ranged from 1×10^4 to 6×10^6 . Also presented, as an appendix, is a theoretical laminar and turbulent heat transfer prediction method based on correlations of exact similarity solutions.

43. **Fitzsimmons, H. D.: Analysis of Hypersonic Pressure and Heat Transfer Tests on Delta Wings with Laminar and Turbulent Boundary Layers. NASA CR-154117, 1966.**

Results are presented of an analysis of slab delta wing pressure and heat transfer data with laminar and turbulent boundary layers. The data were obtained during the X-20 (Dyna-Soar) program from a parametric series of models tested in conventional wind tunnels at Mach numbers of 6, 7, and 8. Shock tunnel data at Mach numbers of 6 and 15 and shock tube data at a Mach number of 2.2 are also presented. All tests were in air. Free stream Reynolds numbers based on leading edge diameter ranged from 1×10^4 to 6×10^6 . Also presented, as an appendix, is a theoretical laminar and turbulent heat transfer prediction method based on correlations of exact similarity solutions.

44. **Giles, H. L.; and Thomas, J. W.: Analysis of Hypersonic Pressure and Heat Transfer Tests on a Flat Plate With a Flap and a Delta Wing With a Body, Elevons, Fins, and Rudders. NASA CR-153994, 1966.**

This report was prepared by The Boeing Company under Contract NAS1-4301, "Analysis of X-20 Parametric Aerothermodynamic Test Data." The work was done for the Langley Research Center of the National Aeronautics and Space Administration with M. H. Bertram acting as NASA contract monitor. Results are presented of an analysis of data taken during the X-20 (Dyna-Soar) program in which hypersonic pressures and heat transfer were measured in regions of boundary layer separation and flow field interference on a sharp flat plate with a flap and a blunt delta wing with a body, elevons, fins, and rudders. The experimental results are compared with empirical and theoretical calculations.

45. **Jones, Robert A.; and Hunt, James L.: Use of Fusible Temperature Indicators for Obtaining Quantitative Aerodynamic Heat Transfer Data. NASA TR R-230, February 1966.**

Some of the methods used for obtaining quantitative aerodynamic heat-transfer data by means of temperature-sensitive coatings are described and discussed. A method whereby data can be obtained on arbitrary shapes without the use of a reference body has been developed. In this method, the heat-transfer coefficients depend only upon the thermal properties of the model material and the time required for a visible phase change of a fusible temperature indicator which is applied to the model as a thin surface coating. The phase change is recovered by motion picture photography, and charts are given which related the time required for the phase change to occur to the heat-transfer coefficient. Data obtained by this method are compared with aerodynamic theory and with data obtained by the conventional thermocouple-calorimeter technique. Several configurations were tested in the Langley Mach 8 variable-density tunnel. The results indicate that this method can be very useful and that accurate data can be obtained.

46. **Raper, James L.; Wright, R. L.; Dillon, J. L.; and Garrett, L. B.: Results of a Flight Test of the Apollo Heat-Shield Material at 28,000 Feet Per Second. NASA TM X-1182, February 1966.**

The fourth in the series of Langley Research Center Scout Reentry Heating Project flight tests (R-4) was performed to evaluate the low-density, filled-epoxy Apollo heat-shield material, Avcoat 5026-39 H/C/G. A standard NASA Scout launch vehicle was used to boost the spacecraft with the materials experiment into a shallow reentry trajectory. The enthalpy corresponding to the entry conditions was 15,700 Btu/lb and the maximum heating rate was 750 Btu/ft²-sec. The primary intent of this report is to present data obtained during the flight test and which will be used in future evaluation of the effectiveness of the ablation material. A detailed description of the spacecraft and experiment sensors, trajectory reconstruction technique, flight data from onboard instrumentation, spacecraft reentry motions, local conditions on the spacecraft, and trajectory environmental conditions is presented. No attempt is made to analyze the flight performance of the Apollo heat-shield material.

47. **Neal, Luther Jr.: A Study of the Pressure, Heat Transfer, and Skin Friction on Sharp and Blunt Flat Plates at Mach 6.8. NASA TN D-3312, April 1966.**

A study of the pressure, heat transfer, and skin-friction distributions over a flat plate with both sharp and blunt leading edges has been conducted at a nominal free-stream Mach number of 6.8, over a range of free-stream Reynolds number per centimeter from about 0.02×10^6 to 0.16×10^6 , and at a wall-to-stagnation temperature ratio of about 0.5. The model consisted of a 25.4-cm wide by 61-cm-long flat plate equipped with interchangeable leading edges having thicknesses of 0.0025, 0.25, and 1.27 cm. The investigation covered all three flow regimes - laminar, transitional, and turbulent.

48. **Holloway, Paul F.; and Morrisette, E. Leon: Roughness Effects on Boundary-Layer Transition for Blunt-Leading-Edge Plates at Mach 6. NASA TN D-3517, August 1966.**

An investigation has been conducted to determine the effects of controlled roughness (spheres) on boundary-layer transition for unswept, blunted plates at a free-stream Mach number of 6. The location of boundary-layer transition was determined by heating-rate distributions downstream of the roughness elements on the centerline of the plates. Experimental data are presented for leading-edge bluntnesses of 0.125 and 0.375 inch. Tests were made for an angle of attack of 0° and for a test unit Reynolds number per foot between 1.2×10^6 and 9.2×10^6 .

49. **Johnson, Charles Borden: A Theoretical and Experimental Study At Mach 8 of Flow Separation of a Flat Plate With Deflected Trailing Edge Flap. NASA TM X-57968, August 1966.**

An experimental investigation was made into the effect of unit Reynolds number, flap angle, and wall temperature on the pressure and flow field of a flap plate model with a trailing-edge flap deflected at angles of 10° , 20° , and 30° relative to the plate surface. These tests were conducted at a nominal Mach number of 8 and a nominal unit Reynolds number (per foot) ranging from 0.22×10^6 to 10.9×10^6 . Also reported are pressure measurements. Schlieren studies, surface oil-flow studies, upstream boundary-layer calculations, and the separated flow region interaction. Results showed good agreement between the experimental and calculated pressures for adiabatic and room-temperature wall conditions. The effect of wall cooling, for transitional separation, showed a reduction in the extent of separation. The peak pressure rise, on the flap, for conditions where the separated shock impinged on the flap, were as high as 58 percent above the inviscid flap pressure.

50. **Bushnell, Dennis M.; and Huffman, Jarrett K.: Investigation of Heat Transfer to the Leading Edge of a 75° Swept Fin With and Without Chordwise Slots and Correlations of Swept Leading Edge Transition Data. NASA TM X-1475, 1967.**

51. Henderson, A., Jr.; and Maddalon, D. V.: Boundary Layer Transition at Hypersonic Mach Numbers. AIAA Paper 67-130, January 1967.

52. Stainback, P. C.: Tests at Mach 8 on Cone, Analyzing Effect of Roughness Elements and Variable Entropy on Transition and Heat Transfer Distribution. AIAA Paper 67-132, January 1967.

53. **Jones, Robert A.; Bushnell, Dennis M.; and Hunt, James L.: Experimental Flow Field and Heat-Transfer Investigation of Several Tension Shell Configurations at a Mach Number of 8. NASA TN D-3800, January 1967.**

An investigation was conducted at a Mach Number of 8 to determine the flow field and pressure and heat-transfer distributions about several bell-shaped configurations called tension shells. This shape provides a possible minimum weight-entry vehicle structure since the shell can be designed as a tension member. Schlieren data were obtained over a range of free-stream Reynolds number based on model base diameter from 0.1×10^6 to 1.5×10^6 and angles of attack from 0° to 31° . Heat-transfer and pressure distributions were obtained at Reynolds numbers based on free-stream conditions and on model base diameter of 0.2×10^6 and 0.5×10^6 , respectively, for a configuration which seemed to have a more stable flow field and higher drag.

54. **Samuels, Richard D.; Peterson, John B., Jr.; and Adcock, Jerry B.: Experimental Investigation of the Turbulent Boundary Layer at a Mach Number of 6 With Heat Transfer at High Reynolds Number. NASA TN D-3858, March 1967.**

An experimental investigation of turbulent boundary-layer skin friction and heat transfer on a hollow cylinder was made at a Mach number of 6 and with two wall temperatures at Reynolds numbers based on distance from the virtual origin from 2.4×10^6 to 28.7×10^6 . Data obtained include total-temperature profiles, velocity profiles, average skin-friction coefficients obtained by the momentum method, and Stanton numbers obtained from use of heat flowmeters.

55. **Bushnell, Dennis M.: Effect of Shock Impingement and Other Factors on Leading Edge Heating Transfer. NASA TN D-4243, 1967.**

56. **Bertram, M. H.; Weinstein, L. M.; Cary, A. M., Jr.; and Arrington, J. P.: Effect of Two-Dimensional Multiple-Wave Distortions on the Heat Transfer to a Wall in Hypersonic Flow. AIAA Paper 67-164, January 1967.**

Extensive tests are reported on the heating distribution on a plate in which a series of shallow protruding waves were embedded. Peak laminar heating on the first wave empirically correlated with results from a previous investigation of single waves. The general flow configuration at the first wave was, in part, amenable to analysis by simple laminar separation concepts even when the boundary-layer thickness was greater than the wave height. The peak heating on succeeding waves in the train was essentially that predicted by assuming that each wave was independent of the other waves and applying the first wave results. With a transitional or turbulent boundary layer, vortices were generally found in the separated flow areas. Turbulent peak heating correlated in much the same manner as laminar results, but with a different Mach number effect. Peak heating on single waves in turbulent attached flow could be predicted, but on waves which caused flow separation, peak heating was significantly higher than predictions. Wave sweep of as much as 70° caused little change in laminar heat transfer. In a transitional-turbulent case, wave sweep gave a significant reduction in peak heating, but the increased heating was still substantial.

57. **Samuels, Richard D.; Perterson, John B., Jr.; and Adcock, Jerry B.: Experimental Investigation of the Turbulent Boundary Layer at a Mach Number of 6 With Heat Transfer at High Reynolds Numbers. NASA TN D-3858, March 1967.**

An experimental investigation of turbulent boundary-layer skin friction and heat transfer on a hollow cylinder was made at a Mach number of 6 and with two wall temperatures at Reynolds numbers based on distance from the virtual origin from 2.4×10^6 to 28.7×10^6 . Data obtained include total-temperature profiles, velocity profiles, average skin-friction coefficients obtained by the momentum method, and Stanton numbers obtained from use of heat flowmeters.

58. **Bushnell, Dennis M.: Local Afterbody Heat Transfer to a Blunt Two-Dimensional Configuration at Mach 8. NASA TM X-60000, June 1967.**

59. **Everhart, Philip E.; and Hamilton, H. Harris: Experimental Investigation of Boundary Layer Transition on a Cooled 7.5° Total-Angle Cone at Mach 10. NASA TN D-4188, October 1967.**

An experimental investigation of boundary-layer transition was conducted on a cooled 7.5° total-angle cone at a free-stream Mach number of 10 and a free-stream unit Reynolds number range of 0.4×10^6 to 2.2×10^6 per foot (1.3×10^6 to 7.2×10^6 per meter). Local unit Reynolds numbers from 0.7×10^6 to 3.1×10^6 per foot (2.3×10^6 to 10.2×10^6 per meter) were obtained at a local Mach number of approximately 9. At a local Mach number of approximately 9, transition Reynolds number was found to be essentially independent of wall temperature for the ratios of wall temperature to adiabatic wall temperature of the investigation. Local transition Reynolds number increases with local unit Reynolds number. The increase in transition Reynolds number (based on the end of transition) with local Mach number that has been noted at moderate supersonic Mach numbers continued to a local Mach number of approximately 9.

60. **Stainback, P. Calvin; and Weinstein, Leonard M.: Aerodynamic Heating In The Vicinity of Corners at Hypersonic Speeds. NASA TN D-4130, November 1967.**

61. **Watts, Joseph D.; and Olinger, Frank V.: Heat Transfer Effects of Surface Protuberances on the X-15 Airplane. NASA TM X-1566, May 1968.**

62. **Keyes, J. Wayne, Goldberg, Theodore J., and Emery, James C.: Turbulent Heat Transfer Associated With Control Surfaces at Mach 6. AIAA Journal, vol. 6, no. 8, August 1968, p. 1612-1613.**

The design of a hypersonic cruise aircraft requires knowledge of heat transfer associated with controls for either attached or separated flow. Of particular importance are the peak heating levels and the effect of Reynolds number on peak heating. This Note presents some experimental turbulent heat-transfer results obtained with wedges mounted on a flat plate and on the tunnel wall and with a delta-wing flap model. This investigation was conducted at Mach 6 at various free-stream Reynolds numbers and an angle of attack of 0° .

63. **Hamilton, H. Harris; and Dearing, J. David: Effect of Hinge-line Bleed on Heat Transfer and Pressure Distribution Over a Wedge-Flap Combination at Mach 10.4. NASA TN D-4686, August 1968.**

An experimental investigation has been conducted to determine the effect of boundary-layer bleed on the heat transfer and pressure distributions over a wedge-flap combination with gap between the wedge and flap. The model was tested, with the wedge at angles of attack of 6.83° and 12.83° and the flap elected up to 30° , at a nominal free-stream Mach number of 10.4 and free-stream Reynolds numbers (based on distance to the flap hinger line) of 0.8×10^6 and 3.6×10^6 .

64. **Bushnell, D. M.; and Weinstein, L. M.: Correlation of Peak Heating for Reattachment of Laminar and Turbulent Separated Flows. AIAA Journal, vol. 5, no. 9, September 1968, pp. 1111-1112.**

65. **Bushnell, Dennis M.; Jones, Robert A.; and Huffman, Jarrett K.: Heat Transfer and Pressure Distributions on a Spherically Blunted 25° Half Angle Cone at Mach 8 and Angles of Attack up to 90° . NASA TN D-4792, 1968.**

Local heat transfer and pressure distributions were measured over the conical portion of the cone. The investigation was conducted at a Mach number of 8 and Reynolds numbers of 0.37 and 1.65×10^6 based on free-stream conditions and model base diameter. The pressure data are generally in good agreement with the semiempirical theory over the entire angle-of-attack range. The heat-transfer data for angles of attack up to 45° are in agreement with predictions using the small-cross-flow theory; whereas, the data at higher angles of attack are fairly well represented by the swept-cylinder theory based on the local cone diameter.

66. **Hunt, James L.; and Jones, Robert A.: Effects of Several Ramp-Fairing, Umbilical and Pad Configurations on Aerodynamic Heating to Apollo Command Module at Mach 8. NASA TM X-1640, September 1968.**

An investigation was conducted in the Langley Mach 8 variable-density hypersonic tunnel to determine the effects of several ramp-fairing--protruding-leeward umbilical configurations with and without simulated wire bundle case, protruding shear pads, recessed shear and compression pads, and a recessed windward afterbody umbilical on heat transfer to the Apollo command module. Tests were made at angles of attack of 18° , 22° , and 26° and free-stream Reynolds numbers, based on model diameters, of 0.76×10^6 and 1.48×10^6 .

67. **Watts, J. D.: Flight Experience with Shock Impingement and Interference Heating on the X-15-2 Research Airplane. NASA TM X-1669, October 1968.**

68. **Dearing, J. David; and Hamilton, H. Harris: Effect of Hinge-Centerline Bleed on Heat Transfer and Pressure Distribution Over a Wedge-Flap Combination at Mach Number 10.4. NASA TN D-4686, November 1968.**

An experimental investigation has been conducted to determine the effect of boundary-layer bleed on the heat transfer and pressure distributions over a wedge-flap combination with a gap between the wedge and flap. The model was tested, with the wedge at angles of attack of 6.83° and 12.83° and the flap deflected up to 30° , at a nominal free-stream Mach number of 10.4 and free-stream Reynolds numbers (based on distance to the flap hinge line) of 0.8×10^6 and 3.6×10^6 .

69. **Dearing, J. David; and Hamilton, H. Harris: Heat-Transfer and Pressure Distributions Inside the Hinge-Line Gap of a Wedge-Flap Combination at Mach Number 10.4. NASA TN D-4911, November 1968.**

Aerodynamic heating inside the gap separating a control flap from the adjacent support structure has been measured at a nominal free-stream Mach number of 10.4. The model, a 10-percent-thick wedge, was fitted with a single flap whose chord was 20 percent of the wedge chord. Angle of attack was varied to change the entrance-to-exit pressure ratio and resulted in local Mach numbers ahead of the gap entrance of from 5.9 to 7.9. Laminar, transitional, and turbulent boundary layers on the wedge near the entrance were obtained by combined variations in local Mach numbers and local Reynolds number.

70. **Whitehead, Allen H., Jr.; and Keyes, J. Wayne: Flow Phenomena and Separation over Delta Wings with Trailing-Edge Flaps at Mach 6. AIAA Journal, vol. 6, no. 12, December 1968, pp. 2380-2387.**

The hypersonic flowfield over highly swept delta wings with various types of separation is investigated. Heat-transfer rates, pressure distributions, and several flow-visualization techniques were used to experimentally examine the flow over a large angle-of-attack range. It was found that, when the boundary-layer type differs across the span prior to separation, complex and unusual flow phenomena develop. When the boundary layer is turbulent over the span of the wing prior to separation, the surface heating and pressures can be estimated by two-dimensional calculations over the wing and flap. At small angles of attack, separation can occur on the lee surface either inboard of, or at the leading edge depending on vehicle geometry and test conditions. By means of several flow-visualization techniques, the separated flow is found to form coiled vortices in a manner similar to that found at subsonic speeds with several significant differences.

71. **Jones, Robert A.; and Hunt, James L.: Measurements of Mutual Interference Heating for a Probe Antenna Mounted on an Apollo Reentry Configuration. NASA TM X-1787, June 1969.**

72. **Hunt, James L.: Heat Transfer to the Windward Side of a Fineness Ratio 2.9 Cylinder with Nearly Spherical Ends at a Mach Number of 6 and Angles of Attack of 50⁰, 70⁰, and 90⁰. NASA TN D-5251, July 1969.**

An investigation was conducted in the Langley 20-inch Mach 6 tunnel to define the heat transfer to a fineness-ratio-2.9 cylinder with nearly spherical ends (1/4-scale model of the graphite lunar module fuel cask (GLFC)). Heat-transfer and schlieren photographs were obtained at a nominal Reynolds number based on model diameter of 1×10^6 at sweep angles of 0⁰, 20⁰, and 40⁰. Beckwith's laminar swept infinite-cylinder theory, small cross-flow theory, and local similarity theory with Libby's three-dimensional stagnation-point theory were used piecewise to predict the laminar heat transfer with acceptable accuracy. Local flat-plate approximations for turbulent heat transfer, calculated by the method of Spalding and Chi, agreed with the data obtained in the turbulent flow regions.

73. **Keyes, J. Wayne: Pressures and Heat Transfer on a 75° Swept Delta Wing With Trailing Edge Flap at Mach 6 and Angles of Attack to 90°. NASA TN D-5418, September 1969.**

An investigation has been made of the effects of deflecting a flap on the field on a 75⁰ swept delta wing with sealed gap for angles of attack from 0⁰ to 90⁰. Pressure and heat-transfer measurements were made with the free-stream Reynolds number (based on wing root chord) varying from 1.3×10^6 to 5.6×10^6 for 0⁰ angle of attack and at 3.4×10^6 for angles greater than 0⁰. Flap deflections from 0⁰ to 40⁰ were investigated. In general, center-line calculations based on existing two-dimensional methods, were in good agreement with the trends of the pressure and heat-transfer distributions if the flow pattern characteristic of the angle-of-attack range and the type of boundary layer are taken into account.

74. **Rumsey, Charles B.; Carter, Howard S.; Hastings, Earl C., Jr.; Raper, James L.; and Zoby, Erenes V.: (U) Initial Results From Flight Measurements of Turbulent Heat Transfer and Boundary-Layer Transition at Local Mach Numbers Near 15 (Reentry F). NASA TM X-1856, September 1969. (CONFIDENTIAL)**

(U) Heat-transfer measurements were made at 20 locations on a 5⁰ half-angle beryllium cone 13-feet long with a graphite tip (initial radius of 0.10 inch) during reentry at a free-stream Mach number of 20 (local Mach number near 15). The angle of attack was less than 1⁰ during the entry from 100,000-feet to 60,000-feet altitude, the period for which heating data are presented.

75. Johnston, P. J.; Cabbage, J. M.; and Weidner, J. P.: **Studies of Engine-Airframe Integration on Hypersonic Aircraft.** AIAA Paper 70-542, May 1970.

Effects of turbo-ramjet exhaust impingement on performance, stability and control of a Mach 6 transport, and thrust vectoring on a scramjet powered Mach 12 vehicle were investigated. Simplified theories for predicting performance benefits from exhaust interference were in agreement with experimental results for the transport. Scramjet engine integration on a Mach 12 lifting body configuration was studied parametrically, and the effects of engine location, thrust deflection, altitude, and Mach number were evaluated on the basis of cruise Breguet factor. Losses due to trim penalties could be minimized by careful consideration of engine integration as well as vehicle static margin.

76. Cary, Aubrey M., Jr.: **Turbulent-Boundary-Layer Heat-Transfer and Transition Measurements With Surface Cooling at Mach 6.** NASA TN D-5863, June 1970.

This experimental investigation shows the effect of wall cooling on turbulent heat-transfer and boundary-layer transition for a sharp-leading-edge flat plate at Mach 6 and Reynolds numbers are as high as 10^7 in the Langley 20-inch hypersonic tunnel. Decreasing the ratio of wall temperature to total temperature had little effect on the turbulent heat-transfer coefficient. The Spalding and Chi method best predicted both the level and trend of the heat-transfer data with wall cooling for the virtual origin located at peak heating and using the modified Karman-Reynolds analogy. The transition Reynolds number increased (30 to 60 percent) when the ratio of wall temperature to total temperature decreased, but no transition reversal occurred.

77. Johnson, Charles B.: **Heat-Transfer Measurements at Mach 8 on a Flat Plate With Deflected Trailing-Edge Flap With Effects of Transition Included.** NASA TN D-5899, July 1970.

An experimental heat-transfer investigation was made on a flat-plate model with a short trailing-edge flap deflected at angles of 10° , 20° , and 30° relative to the plate surface. The tests were conducted at a nominal free-stream Mach number of 8, and the nominal free-stream unit Reynolds number was varied from 0.72×10^6 to 35.8×10^6 per meter. The heat-transfer and schlieren results indicated that transition first occurred for the largest flap angle in the separated region at a unit Reynolds number approximately an order of magnitude lower than that for an undisturbed flat plate. Various theories for laminar flow are applied for separated- and attached-flow conditions, and various theories for turbulent flow are applied to the flow over the wing. The heating near flow reattachment on the flap for transitional separation was as much as two times greater than the theoretical prediction for turbulent flow.

78. Keyes, J. W.; and Watson, R. D.: **Laminar Heating in Interior Corners at Mach 19.** *Journal of Spacecraft and Rockets*, vol. 7, August 1970, pp. 1012-1013.

79. Nagamatsu, H. T.; Pettit, W. T.; and Sheer, R. E., Jr.: **Heat Transfer on a Flat Plate in Helium at Mach Numbers 67.3 and 87.6 and in Hypersonic Corner Flow With Air at Mach Number of 19.** NASA CR-114257, August 1970.

80. Edney, B. E.; Bramlette, T. T.; Ives, J.; Hains, F. D.; and Keyes, J. W.: Theoretical and Experimental Studies of Shock Interference Heating. Bell Aerospace Company Report No. 9500-920-125, October 1970.

81. Whitehead, Allen H., Jr.: Vortices in Separated Flows. AIAA Journal, vol. 8, no. 6, June 1970, pp. 1173-1174.

An experimental study was conducted at Mach 6.8 in the Langley 11-inch tunnel at a unit Reynolds number of 0.33×10^6 per inch on a model similar to that employed by Reding, but with the conical nose replaced by an ogive nose. The schlieren pictures indicate that transition occurs within the separated region just downstream of the cylinder-flare juncture. The transition position and separated-flow characteristics in the present tests are compatible with previous results on geometrically similar models run in the same facility.

82. Cary A. M., Jr.; and Hefner, J. N.: Film Cooling Effectiveness in Hypersonic Turbulent Flow. AIAA Journal, vol. 8, November 1970, pp. 2090-2091.

83. Hunt, James L.; Bushnell, Dennis M.; and Beckwith, Ivan E.: The Compressible Turbulent Boundary Layer on a Blunt Swept Slab With and Without Leading Edge Blowing. NASA TN D-6203, January 1971.

A finite-difference method has been developed to solve the equations for compressible turbulent boundary layers on swept infinite cylinders. Predictions of surface heating are compared with experimental data on a 60°-swept, blunt, slab configuration with and without leading-edge blowing. The test conditions were a free-stream Mach number of 8, a wall-to-total temperature ratio of 0.4, and a range of free-stream Reynolds number, based on leading-edge diameter, from 0.92×10^5 to 9.3×10^5 . Predictions for surface streamlines, Mach number profiles, and boundary-layer thickness have also been compared with experimental data, but for zero blowing only.

84. Hunt, James L.; and Creel, Theodore R., Jr.: Interference Heating and Real-Gas Effects at High Angles of Attack. Space Shuttle Technology Conference, NASA Langley Research Center, NASA TM X-2772, March 1971.

85. Hefner, Jerry N.; Whitehead, A. H., Jr.: Lee Side Heating Investigations. Part 1 - Experimental Lee Side Heating Studies on a Delta Wing Orbiter. NASA Space Shuttle Technology Conference, vol. 1, April 1971, pp. 267-287. (Also available as NASA TM X-2272, March 1976).

86. Cary, A. M., Jr.; and Hefner, J. N.: An Investigation of Film-Cooling Effectiveness and Skin Friction in Hypersonic Turbulent Flow. AIAA Paper 71-599, June 1971.

87. Zoby, Ernest V.; and Rumsey, Charles B.: **Analysis of Free-Flight Laminar, Transitional and Turbulent Heat-Transfer Results at Free-Stream Mach Numbers Near 20 (Reentry F).** NASA TM X-2335, September 1971.

Laminar, transitional, and turbulent heat-transfer data were measured during a reentry flight at a Mach number of 20 on a 5° half-angle cone 13 ft long with an initial nose radius of 0.24 cm 0.1 in. The free-stream Reynolds number increased during the prime data period from 2.1×10^6 to 15.7×10^6 per foot and the ratio of wall to total temperature varied from 0.053 to 0.12. The angle of attack was less than 1° for the prime data period. The experimental laminar and turbulent heating rates are compared with results from existing flat-plate prediction methods. A data correlation of the extent of transition and a simple empirical transition-zone heating correlation are also presented.

88. Zoby, Ernest V.; and Graves, Randolph A., Jr.: **Comparison of Results From Three Prediction Methods With Turbulent Heating Data From Wind-Tunnel and Free-Flight Tests.** NASA TM X-2390, September 1971.

89. Whitehead, Allen H., Jr.; and Bertram, Mitchel H.: **Alleviation of Vortex Induced Heating to the Lee Side of Slender Wings in Hypersonic Flow.** AIAA Journal, vol. 9, no. 9, September 1971, pp. 1870-1872.

The present study was done in the Langley 11-inch Mach 6.8 tunnel employing a sharp-apex delta wing, a rounded (circular-arc) apex delta wing, and hyperbolic and parabolic-planform wings. The leading edges of the models were shaped (<0.075 mm thick) with flat upper surfaces and lower surface level angles (perpendicular to the leading edge) between 18° and 20° . The sharp and rounded-apex delta wing models used in the oil-flow study were swept 75° so that results together with other reported results provide information over at least a small range of sweep for comparison with the hyperbolic and parabolic planform wing results. The delta wings employed in the heating and vapor screen tests were swept 70° .

90. Rao, Dhanvada Madhava: **Hypersonic Lee Surface Heating Alleviation on Delta Wing by Apex Drooping.** AIAA Journal, vol. 9, no. 9, September 1971, pp. 1875-1876.

A brief experimental investigation with a 75° delta wing at 5° incidence was conducted in the Langley 11-inch hypersonic ($M = 6.8$) blowdown tunnel at a unit Reynolds number of 0.2 million/inch (model length Reynolds number of 2 million). Preliminary oil-flow tests showed that apex-drooping successfully eliminated the high shear and vortex-associated feather pattern found on the delta wing without droop. In order to confirm the beneficial effect of vortex-suppression on the centerline heating, experiments on a heat-transfer model using the phase-change paint method were carried out.

91. **Whitehead, Allen H., Jr.; Hefner, Jerry N.; and Rao, D. M.: Lee Surface Vortex Effects Over Configurations in Hypersonic Flow. AIAA Paper 72-77, January 1972.**

The role of vortices in producing intense localized heating to the leeward regions of a variety of configurations is shown and discussed. From experimental results, flow-field models are proposed which account for the interaction of the vortex and the upper surface boundary layer; correlation of the heating is then possible for planar delta wings. Guidelines derived from these studies are offered which reduce the vortex-induced heating by appropriate shaping of configurations such as the space shuttle. Reynolds number is shown to exert a large influence on the magnitude and location of the induced heating.

92. **Hefner, J. N.; and Whitehead, A. H., Jr.: Lee-Side Flow Phenomena on Space Shuttle Configurations at Hypersonic Speeds. Part 2: Studies of Lee-Surface Heating at Hypersonic Mach Numbers. Space Shuttle Aerothermodyn. Technology Conference, vol. 2, February 1972, pp. 451-467.**

Lee surface heating data, obtained at relatively low unit Reynolds numbers at Mach 6 and 19, are discussed with emphasis on the peak heating behavior. Surface pressures measured along the lee meridian of the delta-wing orbiter are presented and analyzed in conjunction with the heating. The effects of nose bluntness and lee surface geometry on the heating are discussed and general guidelines are presented for modifying the lee surface geometry of the shuttle to reduce vortex-induced heating. The application of the wind tunnel results to realistic shuttle flight conditions is discussed.

93. **Hunt, James L.: Heat Transfer to Four Fineness-Ratio 1.6 Hexagonal Prisms with Various Corner Radii at Mach 6. NASA TM X-2446, March 1972.**

An investigation was conducted in the Langley 10-inch Mach 6 tunnel to define the aerodynamic heat transfer to the radioisotope fuel cash (heat source) of the SNAP-19/Pioneer power system. The shape of the SNAP-19/Pioneer heat source is that of a hexagonal prism with flat ends; the fineness ratio, based on maximum (edge to edge) diameter is 1.61. Phase-change-paint heat-transfer data and schlieren photographs were obtained on each possible 1/2-scale entry configurations of the SNAP-19/Pioneer heat source. Tests were conducted over a wide range of attitudes and at nominal Reynolds numbers of 0.33×10^6 , 0.84×10^6 , and 2.2×10^6 based on the length of the unablated configuration.

94. **Eaves, R. H.; Buchanan, T. D.; Warmbrod, J. D.; and Johnson, C. B.: Heat Transfer Investigation of Two Langley Research Center Delta Wing Configurations at a Mach Number of 10.5. NASA CR-120036, March 1972.**

Heat transfer tests for two delta wing configurations were conducted in the hypervelocity wind tunnel. The 24-inch long models were tested at a Mach number of approximately 10.5 and at angles of attack of 20, 40, and 60 degrees over a length Reynolds number range from 5 million to 23 million on 4 May to 4 June 1971. Heat transfer results were obtained from model surface heat gage measurements and thermographic phosphor paint.

95. **Matthews, R. K.; Martindale, W. R.; Warmbrod, J. D.; and Johnson, C. B.: Heat Transfer Investigation of Langley Research Center Transition Models at a Mach Number of 8. NASA CR-120045, March 1972.**

The results are presented of a wind-tunnel test program to determine aerodynamic heat transfer distributions on delta body and straight-body transition models of the space shuttle. Heat transfer rates were determined by the phase-change paint technique on Stycast and RTV models using Tempilag as the surface temperature indicator. The nominal test conditions were: Mach 8, length Reynolds numbers of 5 million and 7.4 million, and angles of attack of 20, 40, and 60 deg. Model details, test conditions, and reduced heat transfer data are included. Data reduction of the phase-change paint photographs was performed by utilizing a new technique.

96. **Birch, S. F.; and Keyes, J. W.: Transition in Compressible Free Shear Layers. Journal of Spacecraft and Rockets, vol. 9, August 1972, pp. 623-624.**

The transition Reynolds number for shear layers produced by interactions between weak and strong shock waves is determined on the basis of experiments performed in a 20-in. (Mach 6) and an 11-in. (Mach 6.9) hypersonic tunnel. A variable angle wedge was used to generate a planar shock wave which interacted with the bow wave of a blunt body. An average value of the transition length (defined as the length along the shear layer from the shock interaction to the point where turbulence became visible on schlieren photographs) was used to determine the transition Reynolds number.

97. **Keyes, J. Wayne; and Morris, Dana J.: Correlations of Peak Heating in Shock Interference Regions at Hypersonic Speeds. Journal of Spacecraft and Rockets, vol. 9, no. 8, August 1972, pp. 621-623.**

This Note presents correlations of measured pressure and heat-transfer peaks for shock/boundary-layer interactions and shear layer attachment on configurations with both two-dimensional and three-dimensional interactions. The peak values were obtained in an extensive investigation of shock interference heating on hemispheres, a 30° included angle wedge, and a 2.54 cm diameter cylindrical leading-edge fin model. The investigation included data for Mach numbers of 6 and 20 over a freestream Reynolds number range from 3.3-25.6 million per meter and specific heat ratios of 1.4 and 1.67. Flat-plate shock generator angles varied from 5° to 25°. Sketches of the types of shock interference patterns as classified by Edney and discussed in the present analysis are shown.

98. **Cary, A. M., Jr.; and Hefner, J. N.: Film Cooling Effectiveness and Skin Friction in Hypersonic Turbulent Flow. AIAA Journal, vol. 10, no. 9, September 1972, pp. 1188-1190.**

99. **Hains, Frank D.; and Keyes, J. Wayne: Shock Interference Heating in Hypersonic Flows. AIAA Journal, vol. 10, no. 11, November 1972, pp. 1441-1447.**

A theoretical and experimental study is made of shock interference heating on a variety of basic body shapes and flow conditions. Measurements of pressure and heat transfer are obtained in four wind tunnels at NASA Langley Research Center. These data cover a Mach number range from 6 to 20, and specific heat ratios from 1.20 to 1.67. Peak heating measurements up to 17 times ordinary stagnation point rates and pressure peaks up to 8 times free-stream pitot pressure are recorded. Numerical results from computer codes developed for each of the six types of interference compare favorably with most of the experimental data. A theoretical parametric study determined the effects of Mach number, specific heat ratio, and impinging shock strength on amplification of pressure and heat transfer. The result of this study show that a knowledge of heating can be important for the design of hypersonic vehicles such as the space shuttle. The problem is intensified by the fact that the particular type of interference and the location of peak heating regions on the vehicle will vary along the flight trajectory.

100. **Hefner, J. N.: Lee-Surface Heating and Flow Phenomena on Space Shuttle Orbiters at Large Angles of Attack and Hypersonic Speeds. NASA TN D-7088, November 1972.**

The lee-surface flow phenomena on a delta wing orbiter and a straight wing orbiter have been investigated at angles of attack between 0 deg and 50 deg at a Mach number of 6. Limited studies of the delta-wing orbiter were conducted at a Mach number of 19. Heat-transfer data, pressure distributions, and oil-flow studies were employed to experimentally examine the nature of the surface flow and the severity of the less-surface heating. The effects of Reynolds number on the flow field and heating were investigated. Problem areas are defined and aread for furtherstudy are recommended.

101. **Henderson, A., Jr.: Aerothermodynamic Technology for Space Shuttle - And Beyond. AIAA Paper 73-59, January 1973.**

102. **Hunt, James L.; and Jones, Robert A.: Heating and Flow-Field Studies on a Straight-Wing Hypersonic Reentry Vehicle at Angles of Attack from 20⁰ and 80⁰ with Simulation of Real-Gas Trends. NASA TN D-7108, March 1973.**

Data are presented from a series of phase change heat-transfer and flow-visualization tests at Mach 7.4, 8, and 10.3 in air, Mach 19.5 in nitrogen, Mach 20.3 in helium, and Mach 6 in tetrafluoromethane (CF₄) on the windward surface of a straight-wing hypersonic reentry configuration for angles of attack from 20° to 80°. The results indicate that: (1) For hypersonic stream Mach numbers, the flow field over the straight-wing configuration is essentially independent of Mach number; (2) The transition Reynolds number decreases with increasing angle of attack; (3) At some "critical" angle of attack, the wing-shock standoff distance is greatly increased and the stagnation line moves downstream from the wing leading edge; (4) The value of the "critical" angle of attack is very sensitive to the flow shock density ratio or effective gamma; and (5) At angles of attack above the "critical" value for all gases, the nondimensional level of heat transfer to the wing is higher for the higher shock-density-ratio flows.

- 103. Keyes, J. Wayne; and Hains, F. D.: Analytical and Experimental Studies of Shock Interference Heating in Hypersonic Flows. NASA TN D-7139, May 1973.**

An analytical and experimental study is presented of the aerodynamic heating resulting from six types of shock interference patterns encountered in high speed flow. Centerline measurements of pressure and heat transfer distributions on basic bodies were obtained in four wind tunnels from Mach numbers from 6 to 20, specific heat ratios from 1.27 to 1.67, and free stream Reynolds numbers from 3-million to 25.6-million per meter. Peak heating and peak pressures up to 17 and 7.5 times stagnation values, respectively, were measured. In general, results obtained from semiempirical methods developed for each of the six types of interference agreed with the experimental peaks.

- 104. Hefner, Jerry N.: Effect of Geometry Variations on Lee-Surface Vortex-Induced Heating for Flat-Bottom Three-Dimensional Bodies at Mach 6. NASA-TN-D-7447, November 1973.**

- 105. Keyes, J. Wayne: Off-Centerline Shock Interference Heating Patterns on Basic Shapes in Hypersonic Flows. NASA TM X-2866, December 1973.**

The results of an experimental study of off-center-line shock-interference heating on basic shapes at hypersonic speeds are presented. The study covered three types of shock-interference patterns over a range of nominal Mach numbers (6 to 20), specific heat ratios (1.40 and 1.67), free-stream Reynolds numbers (8-million to 26-million per meter), and impinging shock strengths. Heat-transfer rates higher than stagnation levels were measured over much of the off-centerline model surface. Peak heating up to 17 times for stagnation heating was measured.

- 106. Hefner, J. N.: Vortex-Induced Heating to a Cone-Cylinder Body at Mach 6. Journal of Spacecraft and Rockets, vol. 11, February 1974, pp. 127-128.**

- 107. Hefner, J. N.; and Whitehead, A. H., Jr.: Vortex-Induced Heating To Cone Flaps at Mach 6. Journal of Spacecraft and Rockets, vol. 11, March 1974, pp. 200-201.**

- 108. Rudy, David H.; and Birch, Stanley F.: Mean Flow Development and Surface Heating for an Attaching Compressible Free Shear Layer. Journal of Spacecraft and Rockets, vol. 11, May 1974, pp. 348-351.**

The present Note presents shear layer velocity profiles before attachment as well as surface heating and pressure data for a turbulent attaching shear layer in a type-III interaction. In the present flow, the shear layer does not stagnate on the model surface since the attachment angle of the shear layer relative to the model surface is small; consequently, reverse flow does not occur in the subsonic regions between the shear layer and the surface.

- 109. Hefner, J. N.; and Cary, A. M., Jr.: Investigation of the Three-Dimensional Turbulent Flow Downstream of Swept Slot Injection in Hypersonic Flow. AIAA Paper 74-679, July 1974.**

- 110. Johnson, Charles B.; and Kaufman, Louis G., II: Interference Heating From Interactions of Shock Waves With Turbulent Boundary Layers at Mach 6. NASA TN D-7649, September 1974.**

This paper presents results of an experimental investigation of interference heating resulting from interactions of shock waves and turbulent boundary layers. Pressure and heat-transfer distributions were measured on a flat plate in the free stream and on the wall of the test section of the Langley Mach 6 high Reynolds number tunnel for Reynolds numbers ranging from 2×10^6 to 400×10^6 . Various incident shock strengths were obtained by varying a wedge-shock generator angle (from 10° to 15°) and by placing a spherical-shock generator at different vertical positions above the instrumented flat plate and tunnel wall.

- 111. DeJarnette, Fred R.; and Hamilton, H. Harris: Aerodynamic Heating on 3-D Bodies Including the Effects of Entropy-Layer Swallowing. Journal of Spacecraft and Rockets, vol. 12, no. 1, January 1975, pp. 5-12.**

A relatively simple method is presented to include the effects of entropy-layer swallowing in a method developed previously for calculating laminar, transitional, and turbulent heating rates on three-dimensional bodies in hypersonic flows. The boundary layer swallows the entropy layer when the boundary layer downstream of the nose region has entrained those streamlines which passed through the nearly normal part of the bow shock wave. The entropy at the edge of the boundary layer is then determined by equating the mass flow inside the boundary layer to that entering part of the bow shock wave. A new inviscid flowfield solution, which is an extension of Maslen's axisymmetric method, is developed to calculate the three-dimensional shock shape and couple the inviscid solution with the viscous solution. The calculated heating rates compare favorably with Mayne's theory and experimental data for blunted circular and elliptical cones at angles of attack.

- 112. Hefner, J. N.; and Cary, A. M., Jr.: Swept-Slot Film-Cooling Effectiveness in Hypersonic Turbulent Flow. Journal of Spacecraft and Rockets, vol. 11, May 1974, pp. 351-352.**

Measurement results are presented for the surface equilibrium temperature downstream of swept slots, with sonic tangential air injection into a thick hypersonic turbulent boundary layer. These results are compared with unswept slot results for cooling effectiveness.

- 113. Lawing, Pierce L.: Analysis of Various Descent Trajectories for a Hypersonic Cruise, Cold Wall Research Airplane. NASA TN D-7860, June 1975.**

The envelope of probable descent operating conditions for a hypersonic air-breathing research airplane has been examined. Descents selected included descents at cruise angle of attack, descents at high-dynamic pressure, descent at high-lift coefficient, descents with turns, and descents with drag brakes. These descents were parametrically exercised and compared from the standpoint of cold-wall (367 K) aircraft heat load. The descent parameters compared with total heat load, peak heating rate, time to landing, time to end of heat pulse, and range. Trends in heat load at a function of cruise Mach number, cruise dynamic pressure, angle-of-attack limitation, pull-up g-load, heading angle, and drag-brake size are presented.

- 114. Johnson, Charles B.; and Darden, Christine M.: Flight Transition Data For Angles of Attack at Mach 22 With Correlations of the Data. NASA TM X-3235, August 1975.**

Boundary-layer transition data for angles of attack from 2.5° to 47° from a flight experiment with a cone that reentered at angles of attack up to 75° are analyzed and their local flow conditions presented. The transition data were obtained from both acoustic and electrostatic sensors. There are 102 transitional and turbulent data points for electrostatic sensors and 16 data points from acoustic sensors. Previously unpublished local flow properties are presented for the 93 transitional and turbulent data points and for the 139 laminar data points all from the electrostatic sensors.

- 115. Keyes, J. W.: Shock Interference Peak Heating Measurements Using Phase Change Coatings. Journal of Spacecraft and Rockets, vol. 13, no. 1, January 1976, pp. 61-62.**

The purpose of this Note is to discuss some of the problems encountered in applying the phase change coating technique and to present new shock interference peak heating data which illustrates these problems. A 5.08-cm diameter hemisphere-cylinder made of silica based epoxy was tested at Mach 6 for freestream Reynolds numbers of 3.3 to 25.6 million per meter. Flat plate shock generate angles varied from 5° to 25° .

- 116. Crawford, D. H.: Shock Interference Heat Transfer To Tank Configurations Mated to a Straight-Wing Space Shuttle Orbiter at Mach Number 10.3 — Investigated in a Langley Hypersonic Wind Tunnel.**

Heat transfer was measured on a space shuttle-tank configuration with no mated orbiter in place and with the orbiter in 10 different mated positions. The orbiter--tank combination was tested at angles of attack of 0° and 5° , at a Mach number 10.3, and at a free-stream Reynolds number of one million based on the length of the tank. Comparison of interference heat transfer with no-interference heat transfer shows that shock interference can increase the heat transfer to the tank by two orders of magnitude along the ray adjacent to the orbiter and can cause high temperature gradients along the tank skin. The relative axial location of the two mated vehicles determine the location of the sharp peaks of extreme heating as well as their magnitude. The other control variables (the angle of attack, the gap, and the cross-section shape) had significant effects that were not as consistent or as extreme.

117. Tannehill, J. C.; Holst, T. L.; Rakich, J. V.; and Keyes, J. W.: Comparison of Two-Dimensional Shock Impingement Computation With Experiment. *AIAA Journal*, vol. 14, no. 4, April 1976, pp. 539-541.

Two-dimensional, viscous blunt-body flow fields with an impinging shock wave were computed using a time-dependent, finite-difference method to solve the complete set of Navier-Stokes equations. These computations were qualitatively compared with existing three-dimensional experiments because of the local of suitable two-dimensional experiments. A series of two-dimensional tests in the Langley 20-inch hypersonic tunnel. In these tests, a planar shock wave was allowed to impinge on the cylindrical leading edge of a fin (shock parallel with centerline of leading edge) results in Type III and Type IV interference patterns. This Note compares the results of this computation with the experiment. In addition, recent numerical experiments showing the effects of grid size and numerical smoothing are presented to better establish the accuracy of the code.

118. Hefner, J. N.: Effect of Geometry Modifications on Effectiveness of Slot Injection in Hypersonic Flow. *AIAA Journal*, vol. 14, June 1976, pp. 817-818.

The study presents wind-tunnel measurements of surface static pressures, equilibrium temperatures, and skin friction downstream of tangential slot injection into a thick turbulent hypersonic boundary layer from two modified slot configurations. The data are compared with results obtained for baseline configurations reported by Cary and Hefer (1970, 1972) to determine whether simple modifications to the slot configuration can produce improved cooling effectiveness and skin friction reduction. The baseline slot configurations are simply modified by thickening the slot lip and by elevating the location of the slot exit above the flat plate. Although the results indicate that simple modifications of the baseline slot configurations can enhance the skin friction reductions obtained with tangential slot injection, slot base drag estimates show that neither of the modifications will lessen the impact of the systems penalties for collecting, ducting, and injecting the slot air.

119. Ferri, A.; Mahrer, A.; and Hefner, J. N.: Investigation of Slot Cooling at High Subsonic Speeds --- for Turbulent Engines. *AIAA Journal*, vol. 14, July 1976, pp. 880-885.

The paper discusses the results of an experimental and numerical investigation of tangential slot injection film cooling with zero pressure gradients in subsonic boundary layers at freestream Mach numbers of 0.4, 0.6, and 0.8. The results are compared with the predictions obtained from a finite-difference boundary-layer program developed by NASA for slot injection into turbulent supersonic boundary layers. The two sets of results are found to compare favorably. The numerical results point to the existence of a unique relation between isothermal effectiveness and adiabatic effectiveness, thereby confirming the existence of similarity conditions for temperature and velocity profiles.

120. Lawing, Pierce L.; and Hunt, James L.: **Aerodynamic Heat Transfer to a Hypersonic Research Aircraft Model (X-24C) at Mach 6.** *Journal of Aircraft*, vol. 13, no. 12, pp. December 1976, pp. 1018-1020.

Heat transfer rates were conducted on a 1/29-scale model of the X-24C-121 configuration in the Langley 6 20-inch Mach tunnel at a Reynolds number of 13×10^6 using the phase-change heat-transfer technique. This note presents some results from the experimental study.

121. Morris, D. J.; and Keyes, J. W.: **Shock Interference Patterns and Heating.** NASA Tech Brief LAR-11497, 1976.

122. Donohoe, John C.; Blackstock, Thomas A.; and Keyes, J. Wayne: **Ablation Measurement on the Gasjet Nose Tip.** *Progress in Astronautics and Aeronautics*, vol. 59, January 1977.

The Gasjet nose tip is proposed to reduce the erosion at the apex of a missile nose flying at hypersonic speeds through a rain storm as it reenters the atmosphere. A forward-facing sonic jet is directed through the tip, introducing a secondary counterflow, which displaces the bow shock and blankets the tip with a protective layer of relatively cool gas. Wind-tunnel experiments are described which proved the validity of measuring nose recession in flight by recording the pressure in the blast tube supplying the Gasjet.

123. Hefner, J. N.; Cary A. M., Jr. and Bushnell, Dennis M.: **Downstream Influence of Swept Slot Injection in Hypersonic Turbulent Flow.** NASA TN D-8506, August 1977.

Results of an experimental and numerical investigation of tangential swept slot injection into a thick turbulent boundary layer at Mach 6 are presented. Film cooling effectiveness, skin friction, and flow structure downstream of the swept slot injection were investigated. The data were compared with that for unswept slots, and it was found that cooling effectiveness and skin friction reductions are not significantly affected by sweeping the slot.

124. Johnson, Charles B.; Taylor, Allan H.; and Weinstein, Irving: **Heat Transfer and Pressure Measurements on a Simulated Elevon Deflected 30° Near Flight Conditions at Mach 7.** NASA TM X-3563, September 1977.

Heat-transfer rates and pressures were obtained on an elevon plate (deflected 30°) and a flat plate upstream of the elevon in the Langley 8-foot high-temperature structures tunnel. The tests duplicated the flight Reynolds number and flight total enthalpy for altitudes of 88,000 ft and 94,000 ft at Mach 7. The heat-transfer and pressure data were used to establish heating and pressure loads that would be experienced during tests in the same facility of thermal protection system panels geometrically similar to the plate configuration of this study. The measured heating was compared with several theoretical predicts, and the closest agreement was obtained with a Schultz-Grunow reference enthalpy method of calculation.

- 125. Watson, R. D.: Wall Cooling Effects on Hypersonic Transition/Turbulent Boundary Layers at High Reynolds Numbers. AIAA Journal, vol. 15, August 1977, pp. 1212-1214.**

Low Reynolds number flow of an ideal gas over a blunt axisymmetric body of large half-angle at small angles of attack is investigated, for the case of laminar hypersonic flow. Time-varying viscous shock layer equations describing the flowfield are obtained from the full Navier-Stokes system by keeping terms to second order in the inverse square root of Re in both viscous and inviscid regions: the equations are valid for moderate to high Re . Drag, skin friction, and heating rates were obtained at small (or zero) angles of attack. Conditions experienced by planetary entry probes during the high-latitude (early) legs of an atmospheric entry trajectory are pertinent to the problem.

- 126. Taylor, Allan H.; Jackson, L. Robert; and Weinstein, Irving: Evaluation of Three Thermal Protection Systems in a Hypersonic High-Heating-Rate Environment Induced by an Elevon Deflected 30° . NASA TM-74077, December 1977.**

Three thermal protection systems proposed for a hypersonic research airplane were subjected to high heating rates in the Langley 8-Foot high-temperature structures tunnel. Metallic heat sink (Lockalloy), reusable surface insulation, and insulator-ablator materials were each tested under similar conditions. The specimens were tested for a 10-second exposure on the windward side of an elevon deflected 30° . The metallic-heat-sink panel exhibited no damage; whereas the reusable-surface-insulation tiles were debonded from the panel and the insulator-ablator panel eroded through its thickness, thus exposing the aluminum structure to the Mach 7 environment.

- 127. Keyes, J. Wayen: Correlation of Turbulent Shear Layer Attachment Peak Heating Near Mach 6. AIAA Journal, vol. 15, no. 12, December 1977, pp. 1821-1823.**

This Note presents a correlation of new turbulent two-dimensional data and peak heating data for attaching free shear layers. The present data were obtained on a 2.54-cm and 5.08-cm-diam cylindrical leading-edge slab 25.4-cm long with widths of 7.62 cm and 10.16 cm, respectively. A sharp leading-edge flat plate (30.48-cm long by 25.4-cm wide) set at 15° and 20° was used to generate plane impinging shocks. The freestream Mach number was 6 with the freestream Reynolds number varying from 3.3×10^6 to $25.6 \times 10^6/m$. Peak heating was measured on the silica-based epoxy models using the phase change coating technique.

128. Talcott, N. A., Jr.; and Hunt, J. L.: Calculated Heating on the Afterbody Nozzle of a Hypersonic Aircraft Concept. *Journal of Aircraft*, vol. 14, no. 12, December 1977, pp. 1247-1248.

Hypersonic scramjet powered aircraft offer attractive potential solutions to future civil and military needs. Current concepts use the entire lower fuselage of the aircraft as part of the propulsion system. The vehicle forebody provides the inlet precompression and the lower aft-end of the vehicle acts as a high-expansion ratio external nozzle. Extreme care must be exercised in designing the external nozzle to assure optimum thrust and lift while minimizing adverse pitching moments that could lead to large aircraft trim penalties or instabilities. Recent developments using substitute gas techniques to simulate high-temperature, real-gas scramjet exhaust flows are reported. This Note presents the results of a parametric analysis to estimate the heating levels on the afterbody nozzle of a typical hypersonic research concept. The aircraft concept examined is approximately 80 ft long with a 18 ft long planar exhaust nozzle incorporating at 20° initial expansion angle.

129. Vahl, Walter A.; and Edwards, C. L. W.: Study of Heat Sink Thermal Protection Systems for Hypersonic Research Aircraft. *AIAA Paper 78-38*, January 1978.

The feasibility of using a single metallic heat sink thermal protection system (TPS) over a projected flight test program for a hypersonic research vehicle was studied using transient thermal analyses and mission performance calculations. Four materials, aluminum, titanium, Lockalloy, and beryllium, as well as several combinations, were evaluated. Influence of trajectory parameters were considered on TPS and mission performance for both the clean vehicle configuration as well as with an experimental scramjet mounted. From this study it was concluded that a metallic heat sink TPS can be effectively employed for a hypersonic research airplane flight envelope which includes dash missions in excess of Mach 8 and 60 seconds of cruise at Mach numbers greater than 6. For best heat-sink TPS match over the flight envelope, Lockalloy and titanium appear to be the most promising candidates.

130. Lawing, Pierce L.: Configuration Heating for a Hypersonic Research Airplane Concept Having a 70° Swept Double-Delta Wing. *NASA TP 1143*, May 1978.

The heating on a candidate hypersonic research airplane configuration has been examined experimentally at Mach 6 by the phase-change-paint technique. The configuration had a double-delta wing with tip fins. Phase-change-paint diagrams give heating data for the model top, side, and bottom, with and without deflected elevons for an angle-of-attack range of 0° to 24° . Nominal Reynolds numbers are on the order of 15×10^6 with supplementary data at length Reynolds number of 4×10^6 , which moves the models from the predominantly turbulent into the predominantly laminar regime. Also, intermediate Reynolds numbers were investigated on the lee side for one angle of attack.

131. Johnson, C. B.; and Kaufman, L. G., II: Heat Transfer Distributions Induced by Elevon Deflections on Swept Wing and Adjacent Surfaces at Mach 6. NASA TM-74045, August 1978.

Surface heat transfer distributions are presented for swept-wing semispan models having trailing-edge elevon-ramp angles of 0° , 10° , 20° , and 30° . The wing sweepback angles are 0° , 50° , and 75° . The models have attachable cylindrical and flat-plate center bodies and various attachment wing-tip fins. The data, obtained for a 0° angle of attack, a free stream Mach number of about 17,000,000, reveal considerably larger regions of elevon induced thermal loads on adjacent surface that would be suggested by fully attached flow analyses.

132. Johnson, C. B.; and Kaufman, L. G. III: High Speed Interference Heating Loads and Pressure Distributions Resulting From Elevon Deflections. AIAA Paper 79-0145, January 1979.

Effects of elevon-induced three-dimensional shock-wave turbulent boundary-layer interactions on hypersonic aircraft surfaces are analyzed. Detailed surface pressure and heating rate distributions obtained on wing-elevon-fuselage models representative of aft portions of hypersonic aircraft are compared with analytical and experimental results from other sources. Examples are presented that may be used to evaluate the adequacy of current theoretical methods for estimating the effects of three-dimensional shock-wave turbulent boundary-layer interactions on hypersonic aircraft surfaces.

133. Kaufman, L. G., II; and Johnson, C. B.: Pressure and Thermal Distributions on Wings and Adjacent Surfaces Induced by Elevon Deflections at Mach 6. NASA TP-1356, March 1979.

Surface pressure distributions and heat transfer distributions were obtained on wing half-models in regions where three dimensional separated flow effects are prominent. Unswept and 50° and 70° swept semispan wings were tested, for trailing-edge-elevon ramp angles of 0° , 10° , 20° , and 30° , with and without cylindrical and flat plate center bodies and with and without various wing-tip plates and fins. The data, obtained for a free stream Mach number of 6 and a wing-root-chord Reynolds number of 18.5 million, reveal considerably larger regions of increased pressure and thermal loads than would be anticipated using nonseparated flow analyses.

134. Lawing, P. L.: A Survey of Heating and Turbulent Boundary Layer Characteristics of Several Hypersonic Research Aircraft Configurations. AIAA Paper 81-1145, June 1981.

Four of the configurations investigated during a proposed NASA Langley hypersonic research aircraft program were selected for phase-change-paint heat-transfer testing and forebody boundary layer pitot surveys. In anticipation of future hypersonic aircraft, both published and unpublished data are results reviewed and presented with the purpose of providing a synoptic heat-transfer data base from the research effort. Engineering heat-transfer predictions are compared with experimental data on both a global and a local basis.

135. Miller, C. G.; Micol, J. R.; Gnoffo, P. A.; and Wilder, S. E.: Heat-Transfer Distributions on Biconics at Incidence in Hypersonic-Hypervelocity He, N₂, Air and CO₂ Flows. AIAA Paper 83-1508, June 1983. (Also available at NASA TM-84667, May 1983.)

Laminar heat transfer rates were measured on spherically blunted, 13°/7° on axis and bent biconics (fore cone bent 7° upward relative to aft cone) at hypersonic hypervelocity flow conditions in the Langley expansion tube. Free-stream velocities from 4.5 to 9.6 km/sec and Mach numbers from 6 to 9 were generated using helium, nitrogen, air, and carbon dioxide test gases, resulting in normal shock density ratios from 4 to 19. Angles of attack, referenced to the axis of the aft cone, was varied from 0° to 20° in 4° increments. The effect of nose bend, angle of attack, and real gas phenomena on heating distributions are presented along with comparisons.

136. Penland, J. A.; Marcum, D. C.; and Stack, S. H.: Wall-Temperature Effects on the Aerodynamics of a Hydrogen-Fueled Transport Concept in Mach 8 Blowdown and Shock Tunnels. NASA TP-2159, July 1983.

Results are presented from two separate tests on the same blended wing-body hydrogen fueled transport model at a Mach number of about 8 and a range of Reynolds numbers (based on theoretical body length) of 0.597×10 to the 6th power to about 156.22×10 to the 6th power. Tests were made in conventional hypersonic blowdown tunnel and a hypersonic shock tunnel at angles at attack of -2 deg to about 8 deg, with an extensive study made at a constant angle of attack of 3 deg. The model boundary-layer flow varied from laminar at the lower Reynolds numbers to predominantly turbulent at the higher Reynolds numbers. Model wall temperatures and stream static temperatures varied widely between the two tests, particularly at the lower Reynolds numbers. These temperature differences resulted in marked variations of the axial-force coefficients between the two tests, due in part to the effects of induced pressure and viscous interaction variations. The normal-force coefficient was essentially independent of Reynolds number. Analysis of results utilized current theoretical computer programs and basic boundary-layer theory.

137. **Miller, C. G.; Gnoffo, P. A.; and Micol, J. R.: Heat-Transfer Distributions for Biconics at Incidence in Hypersonic-Hypervelocity Real-Gas Flow. 14th International Symposium on Shock Tubes and Waves, Sydney, Australia, August 15-18, 1983.**

Laminar heat-transfer distributions were measured on spherically blunted, $13^{\circ}/7^{\circ}$ straight and bent biconics at freestream velocities from 4.5 to 6.9 km/sec and Mach numbers from 6 to 9. The flows were generated in the Langley expansion tube using helium, nitrogen, air, and carbon dioxide; angle of attack, referenced to the axis of the aft cone, was varied from 0° to 20° . The penalty in windward heating to the fore cone due to the 7° nose bend diminished rapidly with increasing angle of attack and was only 10 to 20 percent at the design trim angle of attack of 20° . Leeward heating initially decreased, then increased, with increasing angle of attack. Windward heating rates predicted with a computer code that solves the parabolized Navier-Stokes equations were in good agreement with measurements for helium and air. The study used a 1.9-percent scale model of the proposed generic planetary vehicle and is directly applicable to the orbital transfer vehicle, which incorporates a spherically blunted biconic.

138. **Miller, C. G.; Gnoffo, P. A.; and Wilder, S. E.: Experimental Heating Distributions for Biconics at Incidence in Mach 10 Air and Comparison to Prediction. AIAA Paper 84-2119, August 1984.**

Heating distributions were measured on a 1.9 percent scale model of a generic, aeroassisted vehicle proposed for missions to a number of planets and also a candidate as a moderate L/D Earth orbital transfer vehicle. This vehicle is a spherically blunted, $13^{\circ}/7^{\circ}$ biconic with the fore-cone bent upward 7° to provide self-trim capability; also tested was a straight biconic with the same nose radius and half angles. These measurements were made in the Langley 31-Inch Mach-10 tunnel (formerly known as the continuous flow hypersonic tunnel) at values of the free-stream Reynolds number, based on model length, equal to 0.2 and 0.9 million. The angle of attack, referenced to the aft cone, was varied from 0° to 20° . Heating distributions predicted with a parabolized Navier Stokes (PNS) code are compared to measurement over the present range of Reynolds number and angle of attack. Windward heating was predicted to within 10 percent by the PNS code. Leeward heating distributions were predicted qualitatively for both values of Reynolds number, but quantitative agreement was poorer than on the windward side.

- 139. Johnson, Charles B.: Aerodynamic Heating Rate Distributions Induced By Trailing Edge Controls on Hypersonic Aircraft Configurations at Mach 8. NASA TM-87453, August 1984.**

Aerodynamic surface heating rate distributions in three dimensional shock-wave boundary-layer interaction flow regions are presented for a generic set of model configurations representative of the aft portion of hypersonic aircraft. Heat transfer data were obtained using the phase change coating technique (paint) and, at particular spanwise and streamwise stations for sample cases, by the thin-wall transient temperature technique (thermocouples). Surface oil flow patterns are also shown. The good accuracy of the detailed heat transfer data, as attested in part by their repeatability, is attributable partially to the comparatively high temperature potential of the NASA-Langley Mach 8 aircraft, and should be considered in formulating improvements to empiric analytic methods for calculating heat transfer rate coefficient distributions.

- 140. Miller, Charles G. III: Experimental and Predicted Heating Distributions for Bionics at Incidence in Air at Mach 10. NASA TP-2334, November 1984.**

Heating distributions were measured on a 1.9-percent-scale model of a generic aeroassisted vehicle proposed for missions to a number of planets and for use as a moderate lift-drag ratio Earth orbital transfer vehicle. This vehicle is a spherically blunted, $12.84^\circ/7^\circ$ biconic with the fore-cone bent upward 7° to provide self-trim capability. A straight biconic with the same nose radius and the same half-angles was also tested. The free-stream Reynolds numbers based on model length were equal to about 2×10^5 or 9×10^5 . The angle of attack, referenced to the aft-cone, was varied from 0° to 20° . Heating distributions predicted with a "parabolized" Navier-Stokes (PNS) code are compared with the measurements for the present Reynolds numbers and range of angles of attack. Leeward heating was greatly affected by Reynolds number, with the heating increasing with decreasing Reynolds number for attached flow (low incidence). The opposite was true for separated flow, which occurred when the fore-cone angle of attack exceeded 0.8 times the fore-cone half-angle. Windward heating distributions were predicted to within 10 percent with the PNS code. Leeward heating distributions were predicted qualitatively for both Reynolds numbers, but quantitative agreement was poorer than on the windward side.

- 141. Miller, C. G.; Micol, J. R.; and Gnoffo, P. A.: Laminar Heat-Transfer Distributions on Bionics at Incidence in Hypersonic-Hypervelocity Flows. NASA TM-82213, January 1985.**

- 142. Graves, R. A., Jr.; Hunt, J. L.; and Wander, S. M.: NASA's Hypersonic Fluid and Thermal Physics Program. AIAA Paper 85-0922, June 1985.**

This survey paper gives an overview of NASA's hypersonic fluid and thermal physics programs (recently renamed aerothermodynamics). The purpose is to present the elements of, example results from, and rationale and projection for this program. The program is based on improving the fundamental understanding of aerodynamic and aerothermodynamic flow phenomena over hypersonic vehicles in the continuum, transitional, and rarefied flow regimes. Vehicle design capabilities, computational fluid dynamics, computational chemistry, turbulence modeling, aerothermal loads, orbiter flight data analysis, orbiter experiments, laser photodiagnostics, and facilities are discussed.

- 143. Miller, C. G.; Wilder, S. E.; Gnoffo, P. A.; and Wright, S. A.: Measured and Predicted Vortex-Induced Leeward Heating on a Biconic at Mach 6 and 10. AIAA Paper 85-1061, June 1985.**

Detailed longitudinal and circumferential heating distributions were measured on a spherically blunted, 13 deg/7 deg biconic at angles of attack from 0 deg to 27 deg. The measurements were made in the Langley 20-inch Mach 6 tunnel and 31-inch Mach 10 tunnel to provide heating distributions at Mach 6 and 10 in air. The free-stream Reynolds number based on model length varied from 0.4 to 4.8 million at Mach 6 and 0.3 to 1.3 million at Mach 10. The basic trends in leeward heating are found to be described by a single parameter, the viscous interaction parameter. A decrease in this parameter results in a decrease in leeward heating for attached flow, an increase in heating for separated flow, and a forward movement of the separation region. The parabolized Navier-Stokes code for laminar flow qualitatively predicted the longitudinal and circumferential heating distribution.

- 144. Wells, William L.; MacConochie, Ian O.; Helms, Vernon T. III; and Raney, David: Heating Rate Distributions at Mach 10 on a Circular Body Earth-To-Orbit Transport Vehicle. AIAA Paper 85-0974, June 1985.**

Among the concepts being considered for future Earth-to-Orbit transport vehicles are fully reusable single-stage systems which take off vertically and land horizontally. Because these vehicles carry their own propellant internally, they are much larger than the present Space Shuttle Orbiter. One such single-stage vehicle under study is the circular body configuration which has the advantages of simple structural design and large volume-to-weight ratio. As part of an overall evaluation of this configuration, a series of heat transfer and surface flow tests were conducted. The phase-change paint and oil-flow tests were performed in the Langley 31-inch Mach-10 tunnel at angles of attack from 20 through 40.

145. **Miller, C. G.; Wilder, S. E.; Gnoffo, P. A.; and Wright, S. A.: Measured and Predicted Vortex-Induced Leeward Heating on a Biconic at Mach 6 and 10. AIAA Paper 85-1061, June 1985.**

Detailed longitudinal and circumferential heating distributions were measured on a spherically blunted, $13^{\circ}/7^{\circ}$ biconic at angles of attack from 0° to 27° . The measurements were made in the Langley 20-inch Mach 6 tunnel and 31-inch Mach 10 tunnel to provide heating distributions at Mach 6 and 10 in air. The free-stream Reynolds number based on model length varied from 0.4 to 4.8 million at Mach 6 and 0.3 to 1.3 million at Mach 10. The basic trends in leeward heating are found to be described by a single parameter, the viscous interaction parameter. A decrease in this parameter results in a decrease in heating for separated flow, and a forward movement of the separation region. The parabolized Navier-Stokes code for laminar flow qualitatively predicted the longitudinal and circumferential heating distributions.

146. **Kaufman, L. G., II; and Johnson, C. B.: Aerodynamic Heating Rate Distribution Induced by Trailing Edge Controls on Hypersonic Aircraft Configurations at Mach 8. NASA TM-87453, July 1985.**

Aerodynamic surface heating rate distributions in three dimensional shock-wave boundary-layer interaction flow regions are presented for a generic set of model configurations representative of the aft portion of hypersonic aircraft. Heat transfer data were obtained using the phase change coating technique (paint) and, at particular spanwise and streamwise stations for sample cases, by the thin-wall transient temperature technique (thermocouples). Surface oil flow patterns are also shown. The good accuracy of the detailed heat transfer data, as attested in part by their repeatability, is attributable partially to the comparatively high temperature potential of the NASA-Langley Mach 8 variable density tunnel. The data are well suited to help guide heating analyses of Mach 8 aircraft, and should be considered in formulating improvements to empiric analytic methods for calculating heat transfer rate coefficient distributions.

147. **Kaufman, L. G., II; and Johnson, C. B.: An Overview of Some Investigations of Pressure and Thermal Distribution Induced by Trailing Edge Controls on Hypersonic Aircraft. IUTAM Symposium on Turbulent Shear Layer/Shock Wave Interactions, Palaiseau, France, September 9-12, 1985.**

148. **Hamilton, H. H.; and Weilmuenster, K. J.: Calculation of Convective Heating of Proposed Aeroassist Experiment Vehicle. AIAA Paper 86-1308, June 1986.**

149. Miller, Charles G.; Gnoffo, Peter A.; and Wilder, Sue E.: Measured and Predicted Heating Distributions for Biconics at Mach 10. *Journal of Spacecraft*, vol. 23, no. 3, May-June 1986, pp. 251-258.

Heating distributions were measured on a 1.9% scale model of a generic aeroassisted vehicle proposed for missions to a number of planets and for use as a moderate lift-to-drag Earth orbital transfer vehicle. This vehicle is a spherically blunted, $13^{\circ}/7^{\circ}$ biconic with the fore cone bent upward 7° to provide self-trim capability; also tested was a straight biconic with the same nose radius and half-angles. These measurements were made in the Langley 31-Inch Mach 10 tunnel at values of the freestream Reynolds number, based on model length, equal to 2 and 9×10^5 . The angle of attack, referenced to the aft cone, was varied from 0° to 20° . Heating distributions predicted with a parabolized Navier-Stokes (PNS) code are compared to measurements for the present Reynolds numbers and range of angle of attack. Leeward heating was greatly affected by Reynolds number, with the heating increasing with decreasing Reynolds number for attached flow (low incidence). The opposite was true for separated flow, which occurred when the fore-cone angle of attack exceeded 0.8 times the forecone half-angle. Windward heating distributions were predicted to within 10% by the PNS code. Leeward heating distributions were predicted qualitatively for both values of Reynolds number, but quantitative agreement was poorer than on the windward side.

SUBJECT CATEGORY 4 - CONFIGURATION STUDIES

1. **Becker, John V.: Review of Technology Relating to X-15 Project. NACA Research-Airplane-Committee, Report on Conference on the Progress of the X-15 Project, October 1956.**

From recent studies of the possibilities of flight at very high supersonic speeds there has developed a general consensus that we are on the threshold of an era in which the speed of manned aircraft is likely to increase by an order of magnitude, ultimately exceeding the velocity required for an earth satellite. A primary factor influencing these studies has been the achievement of rocket engines capable of large thrusts in the range needed for boosting long-range manned aircraft. The purpose of this paper is to review the considerations that established in a general way the main features, the performance, and the research missions which appeared feasible in an airplane to be constructed within the specified time limitation.

2. **Ridyard, Herbert W.; Dunning, Robert W.; and Johnston, E. W.: Aerodynamic Characteristics From Wind Tunnel Studies of the X-15 Configuration. NACA Research-Airplane-Committee Report on Conference on the Progress of the X-15 Project, October 1956, pp. 39-56.**

In order to investigate the aerodynamic characteristics of the X-15 research airplane, an exploratory wind-tunnel test program was initiated in January of 1956. Since that time, X-15 models have been tested in eight different facilities through a Mach number range from less than 0.1 to about 6.9. Several variations of the original configurations have been tested. The aerodynamic characteristics of two of the configurations are presented in this paper.

3. **Becker, J. V.: The X-15 Project, Part I - Origins and Research Background. Astronautics and Aerospace Engineering, February 1964, pp. 52-61.**

A history of the research activities which launched the X-15 on its auspicious exploration of hypersonic flight offers valuable lessons now, as the goal of moving manned vehicles into space assumes new dimensions.

4. **Arrington, J. P.; and Johnston, P. J.: Two-Dimensional Separated Flow Studies at High Hypersonic Mach Numbers - An Appraisal. 26th Semi-Annual Meeting of the Supersonic Tunnel Association, Ames Research Center, September 1966.**

5. **Becker, J. V.: The X-15 Program in Retrospect. 3rd Eugene Sanger Memorial Lecture, Jahrestagung der DGLR, Raumfahrt Forschung, Heft 2, vol. 13, March-April 1969, pp. 45-53.**

The principle highlights of the 199-flight program of the X-15 research airplanes are reviewed against a background of the original plans and goals. In these pioneering explorations of hypersonic flight, noteworthy results were obtained in aerospace piloting problems, validation of ground research and developmental testing facilities, and heat transfer in the presence of complex flow fields. The ability of the X-15 system to fly out of the atmosphere for a few minutes of space flight followed by a steep reentry produced additional data of value to the space program. Less tangible but equally important contributions included the focusing and stimulation of countless detailed research and development efforts in many areas, identification of new and unsuspected problems, and the production of new technology in time for urgent but initially unforeseeable applications.

6. **Becker, J. V.: New Approaches to Hypersonic Aircraft. Presented at the 7th Congress of Intl. Council of the Aeronautical Sciences, Rome, Italy, September 14-18, 1970**

The strong interactions between the aerodynamic, structural, and propulsive systems of hypersonic air breathers offer important opportunities for achieved improved vehicles. One of the most promising is the use of hydrogen fuel heat sink to provide cooling of major areas of the airframe. This possibility is explored in some detail, with considerations of the theoretical possibilities, engine designs for minimum cooling, comparative analysis of candidate high-level cooling systems, recent fluid-mechanical studies of slot cooling, structural designs compatible with practical cooling systems, and aerodynamic features made possible in actively cooled vehicles. The results suggest that hypersonic cruise vehicles constructed of titanium alloys are feasible and offer a number of advantages. Further studies of the problems and possibilities of this category of hypersonic vehicles are suggested.

7. **Becker, J. V.; and Kirkham, F. S.: Hypersonic Aircraft - 21st. Century Transports? International Civil Aviation Organization Bulletin, June, 1970.**

The results of a number of mission studies are summarized and the status of hypersonic technology are examined, and the possible pace of technological advances in the future are commented on. The two particularly promising applications for hypersonic aircraft--a long-range transport and a first stage for a reusable launch vehicle--are discussed.

8. **Becker, J. V.; and Kirkham, F. S.: Hypersonic Transperts. NASA SP-292, Conference on Vehicle Technology and Civil Aviation - The Seventies and Beyond, Paper No. 25, November 1971.**

The previous applications papers were concerned with refinements for vehicle types which have already enjoyed massive development. There has been no such large-scale support for hypersonic aircraft, but nevertheless steady progress is being made in the disciplines, as seen from the earlier hypersonics papers in this conference compilation. In the present paper an attempt is made to assess the implications of these disciplinary advances for the hypersonic transport. In projecting some 20 years into the future, it is obviously appropriate that these assessments be uninhibited.

9. **Saland, H.; Fox, H.; and Hoydysh, W.: Studies of Engine-Airframe Integrated Hypersonic Vehicles. NASA CR-112300, 1972.**

A parametric study of an integrated airframe and engine is presented for a hypersonic transport at an altitude of 70,000 feet and a free stream Mach number of 6. The engine considered is a subsonic combustion ramjet using conventional hydrocarbon fuels. The lift-to-drag ratio of the aircraft for two configurations, one with full capture and accelerated flight and the other allowing spillage of the leading shock and in unaccelerated flight, is studied. The parameters varied are the engine efficiencies, the angle of attack, the combustion rates, as well as the captured mass flow. Lift-to-drag ratios on the order of 6.5 are obtained.

10. **Nagel, A. L.; and Becker, J. V.: Key Technology for Airbreathing Hypersonic Aircraft. AIAA Paper 73-58, January 1973.**

Airbreathing hypersonic aircraft can be expected to enter service before the end of the present century. These vehicles are potentially powerful weapons, with flexibility superior to missiles and performance superior to current aircraft. However, improvement in critical ("key") technologies, particularly propulsion and structures, are prerequisite to routine hypersonic flight. One overall requirement is the development of hydrogen as a fuel. Hydrogen is the key to hypersonic flight providing spectacular performance and a large heat sink capability for cooling critical engine and airframe components. Hydrogen SST's could have greatly reduced sonic boom and engine noise. Performance considerations aside, fossil fuel depletion and pollution may well dictate use of hydrogen by the end of the century. This paper reviews recent progress in the key hypersonic technologies, which has been good despite a relatively low priority. Successful hypersonic research engine tests have been made. Active cooling system analysis have shown potential for weight savings, alleviation of structural design problems, and long airframe performance. Adequate progress in the future requires and expanded technology program emphasizing hydrogen usage. A hydrogen fueled hypersonic research airplane is essential, providing critical flight data and operational experience.

11. **Kirkham, F. S.: Hypersonic Transports, Including Airbreathing Launch Vehicles. Working Symposium on Liquid Hydrogen-Fueled Aircraft, NASA Langley Research Center, CN-140228, May 1973, pp. 83-101. (CN-140228)**

In assessing the technology and implications of hypersonic aircraft for transport and space launch applications, one is by necessity projecting 15 to 20 years into the future. It is, of course, impossible to guarantee the eventual emergence of a particular application or to detail aircraft specifications. Just as the jumbo jets were not clearly foreseen 20 years ago, we today cannot predict the exact characteristics and specifications of aircraft 20 years hence. The technology opportunities are apparent however and the history of aeronautical progress is clear, as new technology is developed it is applied.

12. **Kirkham, F. S.; and Driver, C.: Energy Supply and Its Effect on Aircraft of the Future, Part II: Liquid Hydrogen-Fueled Aircraft - Prospects and Design Issues. AIAA Paper No. 73-809, August 1973.**

The performance of hydrogen-fueled commercial aircraft is examined in the subsonic, supersonic, and hypersonic speed regime and compared with JP-fueled systems. Hydrogen aircraft are shown to provide substantial improvements in range and payload fraction as well as to minimize or eliminate many environmental problems. The major elements of a development program required to make hydrogen-fueled aircraft a commercial reality are also outlined and the rationale for and characteristics of both a subsonic demonstrator and a high speed research airplane are described.

13. **Small, W. J.; Fetterman, D. E.; and Bonner, T. F., Jr.: Potential of Hydrogen Fuel for Future Air Transportation Systems. Presented at Intersociety Conference on Transportation, Denver, Colorado, September 23-27, 1973, (ASME Publication 73-ICT-104).**

14. **Henry, John R.; and Anderson, Griffin Y.: Design Considerations for the Airframe-Integrated Scramjet. NASA TM X-2895, December 1973.**

15. **Becker, J. V.; and Kirkham, F. S.: The Hypersonic Transport. International Civil Aviation Organization Bulletin, February 1974.**

Liquid hydrogen, used both coolant and fuel, holds promise for the ultimate in long haul performance and in compatibility with both the environment and energy availability.

16. **Small, W. J.; Fetterman, D. E.; and Bonner, T. F., Jr.: Alternate Fuels for Transportation: Part I - Hydrogen for Aircraft. Published in Mechanical Engineering, May 1974.**

17. **Kirkham, F. S.; Jackson, L. R.; and Weidner, J. P.: The Case for a High-Speed Research Airplane - Results from an In-House Study. AIAA Paper 74-988, August 1974.**

The aircraft conceptual design reported is the result of a feasibility study and does not represent a NASA position on the design requirements for a future research aircraft. Furthermore, several alternative technical solutions exist (e.g. alternative rocket propulsion systems and thermal protection systems) which can potentially fulfill the mission requirements assumed in this report. The conceptual design was constrained by the performance requirements and flight research experiments selected early in the study. (The X-24A and X-24B are example of minimal cost research aircraft although with much more limited objectives than those set forth in the HSRA study.)

18. **Edwards, C. L. W.; Small, W. J.; and Johnston, P. J.: Studies of Scramjet/Airframe Integration Techniques for Hypersonic Aircraft. AIAA Paper 75-58, January 1975.**

19. **Kirkham, F. S.; Jones, R. A.; Buck, M. L.; and Zima, W. P.: Joint USAF/NASA Hypersonic Research Aircraft Study. AIAA Paper 75-1039, August 1975.**

A joint USAF/NASA study has developed a conceptual design for a new high-speed research airplane (X-24C) and identified candidate flight research experiments in the Mach 3 to 6 speed range. Four major categories of high priority research experiments are described as well as the X-24C design concept. The vehicle, a rocket-boosted, delta planform aircraft, is air launched from a B-52 and is capable of forty seconds of rocket cruise at Mach 6 with a research scramjet. Research provisions include a dedicated 10-foot long research experiments section, removable fins and strakes, and provisions for testing integrated airbreathing propulsion systems.

20. **Kirkham, F. S.; Jackson, L. Robert; and Weidner, John P.: Study of a High-Speed Research Airplane. Journal of Aircraft, vol. 12, no. 11, November 1975, pp. 857-863.**

A versatile, low-cost high-speed research airplane (HSRA) is described which is specifically designed to perform meaningful structures and propulsion research for future high-speed aircraft. The HSRA concept can achieve Mach 10 from a B-52 air launch using existing rockets for primary propulsion. The aluminum primary structure and heat sink thermal protection system offers an economical near-term construction. A large-replaceable payload bay is provided to facilitate a wide variety of possible flight experiments. Potential research payloads include turbojet, ramjet and scramjet engines, hydrogen tanks, and hot, insulated, and actively cooled structure.

21. **Hearth, Donald P.; and Press, Albert E.: Hypersonic Technology - Approach to an Expanded Program. Journal of Astronautics and Aeronautics, vol. 14, no. 12, December 1976, pp. 21-37.**

Over the past 20 years, application of hypersonic technology in the United States has been limited almost exclusively to the relatively narrow field of space re-entry systems. So far these systems have been rocket-boosted nonreusable vehicles in the form of either ballistic-missile warheads, reconnaissance-satellite capsules, or manned spacecraft, principally Mercury, Gemini, and Apollo. All the recoverable manned vehicles employed parachute recovery systems and water landings. The progression of re-entry vehicles raised the technology for this class of application to a high level. Now Shuttle development makes the orbital vehicle generally reusable, and research programs are developing technology for fully reusable systems.

22. **Kirkham, F. S.; and Hunt, J. L.: Hypersonic Transport Technology. Acta Astronautica, vol. 4, January-February 1977, pp. 181-199.**

The possibility of developing a hypersonic commercial transport (HST) for the early 21st century is explored in terms of potential performance characteristics and recent advances in propulsion and structures technology. Range-gross weight characteristics indicate that a 200 passenger, Mach 6 aircraft with a range of 9200 km (5000 n.mi) would have a gross take-off weight not too different from that of current wide-body subsonic transports. The low cruise sonic boom overpressures generated by the HST opens the possibility of supersonic overland flight which, if permitted, would greatly increase the attractiveness of this type of aircraft. Recent advances in hypersonic propulsion systems and long-lived hypersonic aircraft structure are also discussed. The airframe-integrated scramjet and the actively-cooled airframe structure are identified as the most promising candidates for the HST and current approaches are described in some detail.

23. **Kirkham, F. S. and Hunt, J. L.: Hypersonic Transport Technology. Acta Astronautica, April 1977.**

24. **Hunt, James L.; Jones, Robert A.; and Woods, William C.: Investigation of Real-Gas and Viscous Effects on the Aerodynamic Characteristics of a 40° Half-Cone With Suggested Correlations for the Shuttle Orbiter. NASA TN D-8418, June 1977.**

The purpose of this paper is to evaluate parameters that might be used to correlate shuttle orbiter aerodynamic data to be used in extrapolating from wind tunnel to flight conditions. Preliminary calculations indicate that the lee-side forces will have an insignificant influence on the aerodynamic characteristics of the orbiter or moderate angle-of-attack entries; therefore, this work is focused on phenomena which have an overriding influence on windward forces, namely, real-gas (equilibrium and nonequilibrium) and viscous-interaction effects.

25. **Jackson, L. Robert; Martin, James A.; and Small, William J.: A Fully Reusable, Horizontal Takeoff Space Transport Concept with Two Small Turbojet Boosters. NASA TM 74087, October 1977.**

Results of a preliminary study of a novel space transport concept are presented. The concept consists of a winged orbiter containing ascent propellants and two small turbojet-powered winged boosters, used for acceleration to supersonic speeds. The concept offers full reuse and horizontal takeoff from numerous existing airports. With current structure and rocket technology, this transport concept has lower gross weight for a selected payload than single-stage-to-orbit concepts, which require structural advancements. Discussion includes alternatives to the baseline space transport concept which improve performance; such as advanced structures, dual-fuel rockets, and lightweight scramjets for the orbiter. Moreover, a concept of using a stretched shuttle orbiter instead of an all-new configuration is discussed for two payload-class vehicles.

26. **Arrington, James P.: Results of Studies on Advanced Winged Space Transportation Systems. Acta Astronautica, vol. 12, no. 6, September 1984, pp. 431-441.**

This paper reviews advanced space transportation studies that have been conducted at the NASA Langley Research Center over the past several years and presents the impact of technology on vehicle size and weight. The focus of this work has been on systems that could become operational in the late 1990's and beyond with the primary emphasis on winged vehicles, both single-stage-to-orbit and two-stage concepts.

27. **Young, J. C.; and Underwood, J. M.: The Aerodynamic Challenges in the Design and Development of the Space Shuttle Orbiter. Space Shuttle Technical Conference, NASA CP No. 2342, January 1985.**

28. **Watts, J. D.; Olinger, F. V.; Weidner, J. P.; Sanders, B. W.; Johnson, S. K.; and Keyes, J. W.: Mach 5 Cruise Aircraft Research. 1985 Langley Symposium on Aerodynamics. NASA CP 2398, April 1985.**

SUBJECT CATEGORY 5 - PROPULSION INTEGRATION - EXPERIMENT

1. **Fetterman, D. E., Jr.:** Effects of Simulated Rocket-Jet Exhaust on Stability and Control of a Research-Type Airplane Configuration at a Mach Number of 6.86. NASA TM X-127, October 1959.
2. **Henry, J. R.; Hasel, L. E.; and Mackley, E. A.:** Hypersonic Inlet Research in the Mach 6 to 7 Range. Presented at SAE National Aeronautics Meeting, New York, New York, April 1960.
3. **Henry, John R.; Mackley, Ernest A.; and Torrence, Marvin G.:** Investigation of the Compression Field and the Flow Distribution in the Throat of a Two-Dimensional Internal-Compression, Mach Number 6.9 Inlet. NASA TM X-605, November 1961.
4. **Andrews, E. H., Jr.; and Agnone, A. M.:** Experimental and Analytical Study of an Inlet Forebody for an Airframe-Integrated Scramjet Concept. NASA TM X-3158.
5. **Kirkham, F. S.; Cabbage, J. M.; Vahl, W. A.; and Small, W. J.:** Results of Hypersonic Airframe-Propulsion System Integration Studies. NASA TN D-4128, October 1967.
6. **Andrews, E. H., Jr.; and Rodgers, R. C.:** A Study of Underexpanded Exhaust Jets of an X-15 Airplane Model and Attached Ramjet Engine Simulator at $M = 6.86$. NASA TM X-1571, May 1968.
7. **Johnston, P.J.; Cabbage, J. M.; and Weidner, J. P.:** Studies of Engine-Airframe Integration on Hypersonic Aircraft. AIAA Paper 70-542, May 1970.

Effects of turbo-ramjet exhaust impingement on performance, stability, and control of a Mach 6 transport, and thrust vectoring on a scramjet powered Mach 12 vehicle were investigated. Simplified theories for predicting performance benefits from exhaust interference were in agreement with experimental results for the transport. Scramjet engine intergration on a Mach 12 lifting body configuration was studied parametrically, and the effects of engine location, thrust deflection, altitude, and Mach number were evaluated on the basis of cruise Breguet factor. Losses due to trim penalties could be minimized by careful consideration of engine integration as well as vehicle static margin.

8. **Cabbage, James M.; and Kirkham, Frank S.:** Investigation of Engine-Exhaust-Airframe Interference on a Cruise Vehicle at Mach 6. NASA TN D-6060, January 1971.

Results from an investigation of the effects of underexpanded engine exhaust flow on the aerodynamic performance and stability of a cruise airplane at Mach 6 are presented. The influence of wing reflex angle and nozzle geometry on exhaust flow interference effects were investigated on a flat-plate model. The experiments were conducted at a free-stream Reynolds number of 17.05×10^6 based on the length of the airplane model over a model angle-of-attack range of 0° to 10° and at nozzle static-pressure ratios from 1 to approximately 4.

9. **Ferri: Hypersonic Engine Airplane Integration. NASA CR-137114, February 1974.**
10. **Johnson, Charles B.; and Marcum, Don C., Jr.: A Parametric Study of Effect of Forebody Shape on Flow Angularity at Mach 8. NASA TN D-7768, November 1974.**

Flow angularity and static pressure measurements have been made on the lower surface of nine forebody models that simulate the bottom forward surface of a hypersonic aircraft. Measurements were made in an area of the forebody that represents the location of an inlet of a scramjet engine. Tests were conducted at a Mach number of 8 for free-stream unit Reynolds numbers per meter of 28×10^6 for angles of attack of 0° , 5° , and 10° and 22×10^6 for angles of attack of 15° and 20° . A parametric variation of the forebody surface investigated the effect of: (1) spanwise curvature, (2) longitudinal curvature, and (3) planform shape on both flow angularity and static pressure distribution. Results of each of the three parametric variations of geometry were compared to those for the same flat-delta forebody.

11. **Small, W. J.; Weidner, J. P.; and Johnston, P. J.: Scramjet Nozzle Design and Analysis as Applied To a Highly Integrated Hypersonic Research Airplane. NASA TM X-71972, November, 1974.**

The configuration and performance of the propulsion system for the hypersonic research vehicle are discussed. A study of the interactions between propulsion and aerodynamics of the highly integrated vehicle was conducted. The hypersonic research vehicle is configured to test the technology of structural and thermal protection systems concepts and the operation of the propulsion system under true flight conditions for most of the hypersonic flight regime. The subjects considered are: (1) research vehicle and scramjet engine configurations to determine fundamental engine sizing constraints, (2) analytical methods for computing airframe and propulsion system components, and (3) characteristics of a candidate nozzle to investigate vehicle stability and acceleration performance.

12. **Edwards, C. L. W.; Small, W. J.; Weidner, J. P.; and Johnston, P. J.: Studies of Scramjet/Airframe Integration Techniques for Hypersonic Aircraft. AIAA Paper 75-78, AIAA 13th Aerospace Sciences Meeting, Pasadena, CA January 20-22, 1975.**

New design and analysis techniques for engine-airframe integration were applied in a recent hypersonic vehicle design study. A new technique was developed to design the vehicle's forebody so that uniform precompressed flow was produced at the inlet entrance. Results are verified with three-dimensional characteristic calculations. Results from a new three-dimensional method for calculation nozzle flows show that the entire lower afterbody of the vehicle can be used as a scramjet exhaust nozzle to achieve efficient, controlled, and stable flight over a wide range of flight conditions.

13. **Andrews, E. H., Jr.; Agnone, A. M.; and Pinckney, S. Z.: Experimental and Analytical Study of an Inlet Forebody for an Airframe-Integrated Scramjet Concept. NASA TM X-3158, January 1975.**

Preliminary analytical and experimental inlet forebody investigations have been conducted at Mach numbers of 6.0 and 8.5. The forebody design concept consisted of a sharp-nosed right circular cone followed by elliptical cross sections. This concept resulted in swept isentropic compression to define the condition of the inviscid flow field developed by the forebody, including flow profiles in the vicinity of cowl leading-edge stations, and the three-dimensional boundary-layer effects. The investigation verified some of the expected differences between the predicted and the experimental results.

14. **Ferri, A.: Hypersonic Engine Airplane Integration. NASA CR-142093, February 1975.**
15. **Torrence, M. G.: Experimental Investigation of a Mach 6 Fixed-Geometry Inlet Featuring a Swept External-Internal Compression Flow Field. NASA TN D-7998, October 1975.**

An investigation of a fixed-geometry, swept external-internal compression inlet was conducted at a Mach number of 6 and a test-section Reynolds number of 1.55×10^7 /meter. The test conditions were constant for all runs with stagnation pressure and temperature at 20 atmospheres and 500 K, respectively. Tests were made at angles of attack of -5° , 0° , 3° , and 5° . Measurements consisted of pitot- and static-pressure surveys in inlet throat, wall static pressures, and surface temperatures. Boundary-layer bleed was provided on the centerbody and on the cowl internal surface. The inlet performance was consistently high with the range of angle of attack tested, with an overall average total pressure recovery of 78 percent and corresponding adiabatic kinetic-energy efficiency of 99 percent. The inlet thrust flow distribution was uniform and the Mach number and pressure level were of the correct magnitude for efficient combustor design. The utilization of swept compression field to meet the starting requirements of a fixed-geometry inlet produced neither flow instability nor a tendency to unstart.

16. **Ferri, A.: Hypersonic Engine Airplane Integration: NASA CR-146256, February 1976.**
17. **Andrews, Earl H., Jr.; and Mackley, Ernest A.: Hypersonic Research Engine/Aerothermodynamic Integration Model, Experimental Results. Volume 2: Mach 6 Performance. NASA TM X-72822, April 1976.**

Computer program performance results of a Mach 6 hypersonic research engine during supersonic and subsonic combustion modes were presented. The combustion mode transition was successfully performed, exit surveys made, and effects of altitude, angle of attack, and inlet spike position were determined during these tests.

18. **Andrews, Earl H., Jr.; and Mackley, Ernest A.: Hypersonic Research Engine/Aerothermodynamic Integration Model: Experimental Results. Volume 3: Mach 7 Component Integration and Performance. NASA TM X-72823, April 1976.**

The NASA Hypersonic Research Engine Project was undertaken to design, develop, and construct a hypersonic research ramjet for high performance and to flight test the developed concept on the X-15-2A airplane over the speed range from Mach 3 to 8. Computer program results are presented here for the Mach 7 component integration and performance tests.

19. **Andrews, Earl H., Jr.; and Mackley, Ernest A.: Hypersonic Research Engine/Aerothermodynamic Integration Model, Experimental Results. Volume 4: Mach 5 Component Integration and Performance. NASA TM X-72824, April 1976.**

Computer program results of a Mach 6 hypersonic research engine during supersonic and subsonic combustion modes were presented. The combustion mode transition was successfully performed, exit surveys made, and effects of altitude, angle of attack, and inlet spike position were determined during these tests.

20. **Hopkins H. B.; Konopka, W.; and Leng, J.: Validation of Scramjet Exhaust Simulation Technique. NASA CR-2688, June 1976.**

Scramjet/airframe integration design philosophy for hypersonic aircraft results in configurations having lower aft surfaces that serve as exhaust nozzles. There is a strong coupling between the exhaust plume and the aerodynamics of the vehicle, making accurate simulation of the engine exhaust mandatory. This report describes the experimental verification of the simulation procedure devised by Grumman Research during a previous NASA contract (NASA-2494). The detonation tube simulator was used to produce an exact simulation of the scramjet exhaust for Mach 8 flight condition. The pressure distributions produced by the "exact" exhaust flow were then duplicated by a cool mixture of Argon and Freon 13B1. Such a substitute gas mixture validated by the detonation tube technique could be used in conventional wind tunnel tests. The results presented in this report show the substitute gas simulation technique to be valid for shockless expansions.

21. **Ferri, A.: Hypersonic Engine Airplane Integration. NASA CR-148911, September 1976.**

22. **Edwards, C. L. W.: A Forebody Design Technique for Highly Integrated Bottom-Mounted Scramjets With Application to a Hypersonic Research Airplane. NASA TN D-8369, December 1976.**

A rapid and simple inviscid technique for designing forebodies which produce uniformly precompressed flows at the inlet entrance for bottom-mounted scramjets has been developed so that geometric constraints resulting from design tradeoffs can be effectively evaluated. The flow fields resulting from several forebody designs generated in support of a conceptual design for a hypersonic research airplane have been analyzed in detail. Three-dimensional characteristics calculations were used to verify the uniform flow conditions. For the designs analyzed, uniform flow was maintained over a wide range of flight conditions (Mach numbers from 4 to 10; angles of attack of 6° and 10°) corresponding to the scramjet operation flight envelope of the research airplane.

23. **Hunt, J. L.; Talcott, N. A., Jr.; and Cubbage, J. M.: Scramjet Exhaust Simulation Technique for Hypersonic Aircraft Nozzle Design and Aerodynamic Tests. AIAA Paper 77-82, January 1977.**

Current design philosophy for scramjet-powered hypersonic aircraft results in configurations with the entire lower surface surface utilized as part of the propulsion system. The lower aft-end of the vehicle acts as a high expansion ratio nozzle. Not only must the external nozzle be designed to extract the maximum possible thrust force from the high energy flow at the combustor exit, but the forces produced by the nozzle must be aligned such that they do not unduly affect aerodynamic balance. The strong coupling between the propulsion system and aerodynamics of the aircraft makes imperative at least a partial simulation of the inlet, exhaust, and external flows of the hydrogen-burning scramjet in conventional facilities for both nozzle formulation and aerodynamic-force data acquisition. Aerodynamic testing methods offer no contemporary approach for such vehicle design requirements. NASA/Langley has pursued an extensive scramjet/airframe integration R&D program for several years and has recently developed a promising technique for simulation of the scramjet results of the research program to develop a scramjet flow simulation technique through the use of substitute gas blends are described in this paper.

24. **Johnson, C. B.; and Lawing, P. L.: Mach 6 Flowfield Survey at the Engine Inlet of a Research Airplane. Journal of Aircraft, vol. 14, no. 4, April 1977, pp. 412-413.**

An important research experiment is the testing of airframe-integrated scramjet which is mounted on the lower surface of the vehicle. During heat-transfer tests conducted on a 1/29-scale epoxy model of the new high-speed research airplane configuration, it was noted that there was a pronounced streak of low heating (cold streak) on the centerline of the bottom surface of the fuselage. It was speculated that the cold streak was caused by inflow on the forebody of the delta planform, and this inflow caused a thickening of the boundary layer along the centerline. Since one of the primary objectives of the research airplane would be the testing of an integrated airbreathing propulsion system, it was feared that the centerline cold streak and associated boundary-layer thickening on the centerline of the forebody would affect the performance of the scramjet engine. Therefore, flowfield surveys were conducted to better define the nature of the vehicle forebody flowfield at the inlet location of the scramjet engine. It is the purpose of this paper to present some results of the flow field surveys.

25. **Cubbage, James M.; and Mercer, Charles E.: Investigation of Effect of Propulsion System Installation and Operation on Aerodynamics of an Airbreathing Hypersonic Airplane at Mach 0.3 to 1.2. NASA TN D-8503, July 1977.**

Results from an investigation of the effects of the operation of a combined turbojet/scramjet propulsion system on the longitudinal aerodynamic characteristics of a 1/60-scale hypersonic airbreathing launch vehicle configuration are presented. Tests were conducted in the Langley 16-foot transonic tunnel at Mach numbers of 0.3 to 1.2 and a corresponding Reynolds number range of 9×10^6 to 18×10^6 . Decomposition products of hydrogen peroxide were used for simulation of the propulsion system exhaust.

26. **Huffman, Jarrett K.; Fox, Charles H., Jr.; and Johnston, Patrick J.: Subsonic Longitudinal Aerodynamic Characteristics and Engine Pressure Distributions for an Aircraft With an Integrated Scramjet Designed for Mach 6 Cruise. NASA TM X-73911, August 1977.**

A 1/10-scale model of a proposed hypersonic aircraft with an integrated scramjet was tested. The investigation took place over a Mach number range from 0.2 to 0.7 and an angle of attack range from 2 deg to approximately 17 deg at a sideslip angle of 0 deg. The primary configuration variables studied were engine location, internal engine geometry, and external engine geometry. The results are presented without analysis.

27. **Johnston, P. J.; Pittman, J. L.; and Huffman, J. K.: Effect of an Integrated Scramjet Installation on the Subsonic Performance of an Aircraft Designed for Mach 6 Cruise. AIAA Paper 77-1230, August 1977.**

Various factors contributing to the high drag caused by the installation of a six-module scramjet engine were determined from wind tunnel tests at Mach numbers from 0.2 to 0.7. Methods for alleviating this drag were also explored. The external exhaust nozzle, required for good cruise performance, was a major contributor. Of the drag produced by the engine modules, a significant fraction was attributable to wall divergence in the combustor. Good drag simulation could be achieved by using a single fuel injection strut having approximately the same cross-sectional area as the three used on the full-scale engine. External exhaust nozzle fences had a small but beneficial effect on maximum L/D and a flap which diverted the flow away from the inlet was effective in decreasing drag but only at low angles of attack.

28. **Talcott, N. A., Jr.; and Hunt, J. L.: Streamtube Analysis of an Hydrogen-Burning Scramjet Exhaust and Simulation Technique. Journal of Aircraft, vol. 14, no. 9, September 1977, pp. 918-920.**

This Note presents the results of a study to determine if this simulation technique can be extended to more complicated flow approaching the complexity of the actual exhaust flow. An analysis is made to determine the state of the flow and the accuracy of the substitute gas simulation in the presence of a shock discontinuity.

29. **Lawing, P. L.; and Johnson, C. B.: Inlet Boundary-Layer Shapes on Four Aircraft Forebodies at Mach 6. Journal of Aircraft, vol. 15, no. 1, January 1978, pp. 62-63.**

The purpose of the present Note is to present the results of boundary-layer shape measurements at the engine inlet (tests made with no engine on forebody) on four different forebody designs. This information should provide a qualitative assessment of future forebody designs which are to be used as engine-inlet-precompression surfaces. The tests were conducted in the Langley 20-inch hypersonic tunnel at Mach 6 and a nominal freestream Reynolds number of $30.5 \times 10^6 \text{ m}^{-1}$ ($9.3 \times 10^6 \text{ ft}^{-1}$).

30. **Johnston, P. J.; Pittman, J. L.; and Huffman, J. K.: Integrated Scramjet Installation Effect on the Subsonic Performance of a Hypersonic Aircraft. Journal of Aircraft, vol. 15, no. 6, June 1978, pp. 326-332. (Also available as NASA TM X-73911.)**

Various factors contribution to the high drag caused by the installation of a six-module scramjet engine were determined from wind-tunnel tests at Mach numbers from 0.2 to 0.7. Methods for alleviating this drag were also explored. The external exhaust nozzle, required for good cruise performance, was a major contributor. Of the drag produced by the engine modules, a significant fraction was attributable to wall divergence in the combustor. Good drag simulation could be achieved by using a single fuel injection strut having approximately the same cross-sectional area as the three used on the full-scale engine. External exhaust nozzle fences had a small but beneficial effect on maximum L/D, and a flap that diverted the flow away from the inlet was effective in decreasing drag but only at low angles of attack.

36. **Vahl, Walter A.: Experimental Determination of Flow-Interference Effects of Wing-Mounted, Two-Dimensional, Full-Capture Propulsion Nacelles in Close Proximity to a Vehicle Body at a Mach Number of 6. NASA TM-83287, May 1982.**

Experimental tests have been conducted to determine possible aerodynamic interference effects due to the lateral positioning of two-dimensional propulsion nacelles mounted on a wing surface in close proximity to a vehicle body. The tests were conducted at a Mach number of 6 and a Reynolds number of 7×10^6 per foot. The angle-of-attack range for force tests was -9° to 9° . The model configuration consisted of combinations of rectangular and trapezoidal cross-section bodies with a wing swept of 65° and a rectangular planform study. A pair of two-dimensional, flow-through propulsion nacelles simulated full-capture inlet operation.

37. **Johnston, P. J.; and Hunt, J. L.: Mach 6 Flow-Field and Boundary-Layer Surveys Beneath the Forebody of an Airbreathing Missile. AIAA Paper 84-0233, January 1984.**

A wind-tunnel investigation has been conducted to determine the flow characteristics of the shock layer beneath the forebody of a hypersonic, airbreathing missile incorporating an aft-mounted inlet. In the inviscid part of the flow field, the measured parameters were in agreement with those predicted by a three-dimensional Method of Characteristics theory. At test conditions matching full-scale Mach and Reynolds numbers at an altitude for maximum L/D cruise, boundary-layer transition occurred downstream of the inlet fact. While this means that a full size vehicle would have a more easily separated boundary layer, it is noted that the Reynolds number discrepancy between wind tunnel and actual flight conditions may result in a movement of the transition point to somewhere upstream of the inlet.

38. **Vahl, Walter A.: Aerodynamic Effects of Inlet Spillage of a Hypersonic Configuration at Mach Numbers of 1.50 to 2.85. (U) NASA TM 85736, February 1984. (CONFIDENTIAL)**

(U) Experimental tests have been conducted to determine the aerodynamic characteristics of a representative hypersonic configuration when operating at off-design low supersonic Mach numbers with unstarted two-dimensional inlets spilling large quantities of air. The tests were conducted over a Mach number range from 1.50 to 2.85 at a Reynolds number of 2×10^6 per foot. The angle of attack was varied from -3° to 12° . Two nacelle cowl shapes were tested. One shape represented a cowl at the hypersonic cruise design position and remained fixed in this position over the entire flight Mach number range, and the other shape represented a movable cowl capable of matching the much smaller off-design engine airflow requirements.

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SUBJECT CATEGORY 6 - PROPULSION INTEGRATION - STUDY

- 1. Bertram, M. H.; Fetterman, D. E.; and Henry, J. R.: The Aerodynamics of Hypersonic Cruising and Boost Vehicles. NASA-University Conference on the Science and Technology of Space Exploration, Chicago, Illinois. NASA SP-11, Nov. 1962.**
- 2. Johnston, P. J.; Cabbage, J. M.; and Weidner, J. P.: Studies of Engine-Airframe Integration on Hypersonic Aircraft. AIAA Journal of Aircraft, July 1971, pp. 495-501.**

Effects of turbo-ramjet exhaust impingement on performance, stability, and control of a Mach 6 transport, and thrust vectoring on a scramjet-powered Mach 12 vehicle were investigated. Simplified theories for predicting performance benefits from exhaust interference were in agreement with experimental results for the transport. Scramjet engine integration on a Mach 12 lifting body configuration was studied parametrically, and the effects of engine location, thrust deflection, altitude, and Mach number were evaluated on the basis of cruise Breguet factor. Losses due to trim penalties could be minimized by careful consideration of engine integration as well as vehicle static margin.

- 3. Henry, J. R.; and Anderson, G. Y.: Design Considerations for the Airframe-Integrated Scramjet. NASA TM X-2895, December 1973.**

Research programs at the NASA Langley Research Center on the development of airframe-integrated scramjet concepts (supersonic combustion ramjet) are reviewed briefly. The design and performance of a specific scramjet configuration are examined analytically by use of recently developed and substantiated techniques on boundary-layer development, heat transfer, fuel-air mixing, heat release rates, and engine-cycle analysis. These studies indicate that the fixed geometry scramjet module will provide practical levels of thrust performance with low cooling requirements. Areas which need particular emphasis in further development work are the combustor design for low speeds and the integrated nozzle design.

- 4. Oman, R. A.; Foreman, K. M.; Leng, J.; and Hopkins, H. B.: Simulation of Hypersonic Scramjet Exhaust. Grumman Aerospace Corporation Report RE-473, Final Report Prepared under Contract NAS1-12553, January 1974.**
- 5. Edwards, C. L. W.: A Forebody Design Technique for Highly Integrated Bottom-Mounted Scramjets with Application to a Hypersonic Research Airplane. NASA TM X-71971, November 1974.**

A rapid and simple inviscid technique for designing forebodies which produce uniformly precompressed flows at the inlet entrance for bottom-mounted scramjets has been developed so that geometric constraints resulting from design trade-offs can be effectively evaluated. The flow fields resulting from several forebody designs generated in support of a hypersonic research airplane conceptual design study have been analyzed in detail with three-dimensional characteristics calculations to verify the uniform flow conditions. For the designs analyzed, uniform flow is maintained over a wide range of flight conditions corresponding to scramjet operation flight envelope of the research airplane.

6. **Oman, R. A.; Foreman, K. M.; Leng, J.; and Hopkins, H. B.: Simulation of Hypersonic Scramjet Exhaust --- Pressure Distribution on Afterbody/Nozzle Sections of Vehicle. NASA CR-2494, March 1975.**

A plan and some preliminary analysis for the accurate simulation of pressure distributions of the afterbody/nozzle portions of a hypersonic scramjet vehicle are described. The objectives fulfilled were to establish the standards of similitude for a hydrogen/air scramjet exhaust interacting with a vehicle afterbody, determine an experimental technique for validation of the procedures that will be used in conventional wind tunnel facilities, suggest a program of experiments for proof of the concept, and explore any unresolved problems in the proposed simulation procedures. It is shown that true enthalpy, Reynolds number, and nearly exact chemistry can be provided in the exhaust flow for the flight regime from Mach 4 to 10 by a detonation tube simulation. A detailed discussion of the required similarity parameters leads to the conclusion that substitute gases can be used as the simulated exhaust gas in a wind tunnel to achieve the correct interaction forces and moments.

7. **Hopkins, H. B.; Konopka, W.; and Lens, J.: Validation of Scramjet Exhaust Simulation Technique. NASA CR-2688, June 1976.**

Scramjet/airframe integration design philosophy for hypersonic aircraft results in configuration having lower aft surfaces that serve as exhaust nozzles. There is a strong coupling between the exhaust plume and the aerodynamics of the vehicle, making accurate simulation of the engine exhaust mandatory. The experimental verification of the simulation procedure is described. The detonation tube simulator was used to produce an exact simulation of the scramjet exhaust for a Mach 8 flight condition. The pressure distributions produced by the exact exhaust flow were then duplicated by a cool mixture Argon and Freon 1381. Such a substitute gas mixture validated by the detonation tube technique could be used in conventional wind tunnel test. The results presented show the substitute gas simulation technique to be valid for shockless expansions.

8. **Weidner, J. P.; Small, W. J.; and Penland, J. A.: Scramjet Integration on Hypersonic Research Airplane Concepts. AIAA Paper No. 76-755, July 1976.**

9. **Small, W. J.; Weidner, J. P.; and Johnston, P. J.: Scramjet Nozzle Design and Analysis As Applied to a Highly Integrated Hypersonic Research Airplane. NASA TN D-8334, November 1976.**

Engine-nozzle airframe integration at hypersonic speeds was conducted by using a high-speed research aircraft concept as a focus. Recently developed techniques for analysis of scramjet-nozzle exhaust flows provide a realistic analysis of complex forces resulting from the engine-nozzle airframe coupling. By properly integrating the engine-nozzle propulsive system with the airframe, efficient, controlled and stable flight results over a wide speed range.

10. **Weidner, J. P.; Small, W. J.; and Penland J. A.: Scramjet Integration on Hypersonic Research Airplane Concepts. Journal of Aircraft, Vol. 14, No. 5, May 1977.**

Several rocket boosted research airplane concepts were evaluated with a research scramjet engine to determine their potential to provide research on critical aspects of airframe-integrated hypersonic systems. Extensive calculations to determine the force and moment contributions of the scramjet inlet, combustor, nozzle, and airframe were conducted to evaluate the overall performance of the combined engine/airframe system at hypersonic speeds. Results of both wind-tunnel tests and analysis indicate that it is possible to develop a research airplane configuration that will cruise at hypersonic speed on scramjet power alone and also will have acceptable low-speed aerodynamic characteristics for landing.

11. **Hartill, W. R.; Goebel, T. P.; and Van Camp, T. P.: Study of Hypersonic Propulsion/Airframe Integration Technology. NASA CR-145321, January 1978.**

This report describes an assessment of current and potential ground facilities, and analysis and flight test techniques for establishing a hypersonic propulsion/airframe integration technology base. A Mach 6 cruise prototype aircraft incorporating NASA Langley Research Center integrated scramjet engines was considered the baseline configuration, and the assessment focused on the aerodynamic and configuration aspects of the integration technology. The study describes the key technology milestones that must be met to permit a decision on development of a prototype vehicle, and defines risk levels for these milestones. Capabilities and limitations of analysis techniques, current and potential ground test facilities, and flight test techniques are described in terms of the milestones and risk levels.

12. **Morris, R. E.; and Brewer, G. D.: Hypersonic Cruise Aircraft Propulsion Integration Study. NASA CR-158926-2, 1979.**

Conceptual vehicle configuration and propulsion approach for a Mach 6 transport aircraft capable of carrying 200 passengers 9260 km was investigated. Wing tunnel test data for various hypersonic transport configurations were examined. Candidates for baseline reference vehicles were selected. An explanation of technical methods which were used and configuration details which were significant in the final vehicle concept are given.

13. **Weidner, J. P.: Conceptual Study of a Turbojet/Ramjet Inlet. NASA TM 80141, September 1979.**

An inlet concept for separate turbojet and ramjet engines was defined and compared with an equivalent inlet for a wraparound turboramjet engine. The comparison was made for a typical high-altitude hypersonic cruise vehicle where the turbojet inlet capture area was required to be half as large as the ramjet inlet capture area at cruise. Results of the study suggest the use of a shorter nacelle having substantially lower cooling requirements at cruise for the inlet concept for separate turbojet and ramjet engines. In addition, the separate engine concept better isolates the turbojet from the ramjet, requires no special close-off mechanisms within the turbojet, and avoids the circumferential heat load imposed by a wrap around ramjet, but it does not require more variable geometry.

14. Weidner, J. P.: Propulsion/Airframe Integration Considerations for High Altitude Hypersonic Cruise Vehicles. Pasadena, California, January 1980. AIAA Paper No. 80-0111.
15. Baker, A. A.: Propulsion System Concepts for Supersonic/Hypersonic Aircraft (U). NASA CR 159326, August 1980. SECRET
16. Vahl, W. A.; and Weidner, J. P.: A Preliminary Assessment of the Impact of 2-D Exhaust-Nozzle Geometry on the Cruise Range of a Hypersonic Aircraft with Top-Mounted Ramjet Propulsion. NASA TM 81841, September 1980.

A theoretical study of full-length and shortened, two-dimensional, isentropic, exhaust nozzles integrated with top-mounted ramjet-propulsion nacelles has been conducted. Both symmetric and asymmetric contoured nozzles with a range of angular orientations were considered. Performance comparisons to determine optimum installations for a representative hypersonic vehicle at Mach 5 cruise conditions are presented on the basis of cruise range, propulsive specific impulse, inlet area requirements, and overall lift-drag ratio. Aerodynamic trim was not considered. The effect of approximating the nozzle internal contours with planar surfaces and the determination of viscous- and frozen-flow effects are also presented.

17. Baker, A.A.; and Morris, R. E.: Propulsion Integration Study of a Turbofan Ramjet Powered Aircraft (U). NASA CR 165678, June 1981. SECRET
18. Matranga, G. J.; and King, A. D.: Design Study of Two Turboramjet Propulsion Concepts for a Mach 5 Penetrator Aircraft (U). NASA CR 172145, March 1984. SECRET
19. Perkins, E. W.; Rose, W. C.; and Horie, G.: Design of a Mach 5 Inlet System Model. Lockheed-California Company, NASA CR 3830, August 1984. CONFIDENTIAL
20. Cassidy, M. D.: Performance Sensitivities of a Mach 5 Penetrator Aircraft Concept. (U) NASA CR 3932, September 1985. SECRET

SUBJECT CATEGORY 7 - ANALYSIS METHODS

1. **Becker, J. V.: Results of Recent Hypersonic Unsteady Flow Research at the Langley Aeronautical Laboratory. Aviation Week, July 31, 1950, pp. 22, 24.**
2. **Bertram, Mitchel H.: An Approximate Method for Determining the Displacement Effects and Viscous Drag of Laminar Boundary Layers in Two-Dimensional Flow. NASA TN 2773, September 1952.**
3. **Rainey, R. W.: Working Charts for Rapid Prediction of Force and Pressure Coefficients on Arbitrary Bodies of Revolution by Use of Newtonian Concepts. NASA TN D-176, December 1959.**

A simplified method, with working charts, is presented so that the loading, or pressure, distributions and aerodynamic characteristics of arbitrary bodies of revolution may be obtained by use of Newtonian concepts.

4. **Bertram, Mitchel H.; and Blackstock, T. A.: Some Simple Solutions to the Problem of Predicting Boundary-Layer Self-Induced Pressures. NASA TN D-798, April 1961.**

Simplified theoretical approaches are shown, based on hypersonic similarity boundary-layer theory, which allow reasonably accurate estimates to be made of the surface pressures on plates on which viscous effects are important. The consideration of viscous effects includes the cases where curved surfaces, stream pressure gradients, and leading-edge bluntness are important factors.

5. **Bertram, Mitchel H.: Calculations of Compressible Average Turbulent Skin Friction. NASA TR R-123, 1962.**
6. **Fetterman, David E.: A Method for Predicting the Normal-Force Characteristics of Delta Wings at Angles of Attack from 0° to 90°. NASA TM X-757, March 1963.**

A method for predicting the normal-force characteristics of delta wings of various leading-edge sweeps and radii is proposed. The method utilizes a system of three basic equations, each of which is confined to a particular angle-of-attack range definable in terms of leading-edge sweep and free-stream Mach number, and it can be applied to both flat-plate wings and those with low to moderate amounts of dihedral. Comparisons of the results with experimental data from representative configurations at supersonic and hypersonic Mach numbers indicate the method to be more generally applicable than existing theories and to have useful application at Mach numbers of greater than 3. A modification allows the prediction of the average center-line pressure coefficients for flat-plate delta wings of various sweeps with better accuracy than methods now in existence. An extension to account for the effects of arbitrary specific-heat ratio is included in the appendix.

7. **Stallings, Robert L.; Jr., Burbank, P. B.; and Howell, Dorothy T.: Heat Transfer and Pressure Measurements on Delta Wings at Mach Numbers of 3.51 and 4.65 and Angles of Attack From -45° to 45° . NASA TN D-2387, August 1964.**

Throughout the angle-of-attack range for which the stagnation line remained on the wing cylindrical leading edge, the pressure distributions on the leading edges of the models through 60° expansion leeward of the stagnation line were predicted by both Newtonian flow and a modified version of the compressible Bernoulli equation based on normal Mach number components. For expansion angles from 60° to 90° , the experimental data were in agreement with the compressible Bernoulli equation. For angles of attack for which the stagnation line shifts from the geometrical wing leading edges to the ridge line, the pressure distributions on the wing leading edges (now trailing edges) were approximated only by the Newtonian flow.

8. **Ashby, George C., Jr.; and Goldberg, Theodore: Application of Generalized Newtonian Theory to Three-Dimensional Sharp-Nose Shock-Detached Bodies at Mach 6 for Angles of Attack Up to 25° . NASA TN D-2550, 1965.**

From measured pressure distributions the maximum pressure coefficient was found to occur at apex deflection angles of 61° , 65° , and 69° for the conical, circular-arc, and parabolic-arc bodies, respectively. The generalized form of Newtonian theory can be used to predict pressures on most curved three-dimensional bodies by the method shown herein. The pressure distributions on aerodynamically blunt cones, flat plates, and aerodynamically blunt wedges at corresponding apex or leading-edge deflection angles are essentially coincident aft of the more rearward maximum-pressure point.

9. **Zoby, E. Vincent; and Sullivan, E. M.: Correlation of Free-Flight Turbulent Heat Transfer Data From Axisymmetric Bodies With Compressible Flat-Plate Relationships. NASA TN D-3802, January 1967.**

Published experimental turbulent heat-transfer data obtained over a range of free-flight conditions and body shapes were compared with calculated turbulent flat-plate values. The calculated values were evaluated by use of a modified Reynolds analogy and the skin-friction relationships of Blasius or Schultz-Grunow with compressibility effects accounted for by evaluating the flow properties on reference conditions. For reference Reynolds numbers less than 10^7 , the calculated heating rates based on either of the two methods correlated well with the experimental data. For reference Reynolds numbers greater than 10^7 and less than 8×10^7 , the calculated heating rates based on the Schultz-Grunow relation compare better with the available experimental data.

10. **Crawford, D. H.: Applications of Similar Solutions for Calculations of Laminar Boundary-Layer Characteristics in the Presence of a Pressure Gradient. AIAA Journal, vol. 5, April 1967, pp. 799-801.**

The present note expresses the heat-transfer, skin-friction, and boundary-layer thickness parameters directly in terms of the pressure distribution and length of the boundary layer. Furthermore, these results are evolved directly from the similar calculations without the use of the Li-Nagamatsu similarity assumption and, in this final form, reduce the equations of the method of Bertram when the hypersonic approximations are applied.

11. **Neal, Luther Jr.; and Bertram, Mitchel H.: Turbulent-Skin-Friction and Heat-Transfer Charts Adapted from the Spalding and Chi Method. NASA TN D-3969, May 1967.**

Charts are presented which allow a rapid determination of local and average skin friction and heat transfer on plates in air. The charts cover a Mach number range from 0 to 20, a Reynolds number range from 10^5 to 10^9 in decade increments of the exponent, and a wall-temperature--stagnation-temperature ratio range of 0.1 to the adiabatic-wall case.

12. **Zoby, E. Vincent: Approximate Relations for Laminar Heat-Transfer and Shear Stress Functions in Equilibrium Dissociated Air. NASA TN D-4484, April 1968.**

Simple approximate equations have been developed for computing the normal derivatives of enthalpy and velocity evaluated at the surface of a flat plate or cone and of a blunt axisymmetric body. These approximate equations were developed by correlating exact similar solutions to the laminar boundary-layer equations for equilibrium dissociated air. The results of these approximate equations represent solutions within ± 10 percent.

13. **Beckwith, I. E.; and Bushnell, D. M.: Calculation of Mean and Fluctuating Properties of Incompressible Turbulent Boundary Layer. NASA TM X-61170, 1968.**

14. **Beckwith, Ivan E.; and Bushnell, Dennis M.: Detailed Description and Results for Computing Mean and Fluctuating Quantities in Turbulent Boundary Layers. NASA TN D-4715, October 1968.**

15. **Bushnell, Dennis M.; Johnson, Charles B.; Harvey, William D.; and Feller, William V.: Comparison of Prediction Methods and Studies of Relaxation in Hypersonic Turbulent Nozzle-Wall Boundary Layers. NASA TN D-5433, September 1969.**

Recent turbulent boundary-layer measurements on axisymmetric nozzle walls at Mach numbers of 6.8 and 19 are presented and compared with the predictions of two theoretical methods. One of the methods was of the integral type; the other was a finite-difference solution to the governing differential equations. The integral method gave good predictions of the data at Mach 6 and 8 and a reasonable prediction at Mach 19. Results for the finite-difference approach were available only at Mach 8, for which good agreement with data was obtained. A survey of boundary-layer data indicates that the variation of total temperature with velocity for flat-plate type of flows can be represented within the scatter of the data by the linear Crocco relation, whereas for nozzle-wall flows, this relation is generally more nearly quadratic. A finite-difference solution for the Mach 8 nozzle-wall boundary layer resulted in a quadratic-type temperature-velocity relation in the strong favorable pressure-gradient region downstream of the throat. However, near the nozzle exit the theoretical solution tended to approach the linear variation of Crocco. Apparently, the observed quadratic variation in nozzle boundary layers can be at least partly accounted for by the pressure-gradient history of the flow.

16. **Hunt, James L.; Bushnell, Dennis M.; and Beckwith, Ivan E.: Finite-Difference Analysis of the Compressible Turbulent Boundary Layers on a Blunt Swept Slab with Leading Edge Blowing. NASA SP-228, Symposium on Analytic Methods in Aircraft Aerodynamics, October 1969.**
17. **Cary, Aubrey M., Jr.: Summary of Available Information on Reynolds Analogy for Zero-Pressure-Gradient, Compressible Turbulent-Boundary-Layer Flow. NASA TN D-5560, January 1970.**
18. **Goldberg, T. J.; and Hefner, J. N.: Starting Criterion for Hypersonic Inlets. Journal of Aircraft, vol. 7, June 1970, pp. 275-277.**

19. **Bushnell, D. M.; and Beckwith, L. E.: Calculations of Nonequilibrium Hypersonic Turbulent Boundary Layers and Comparisons with Experimental Data. AIAA Journal, vol. 8, no. 8, August 1970, pp. 1462-1469.**

The compressible turbulent boundary-layer equations are solved by a finite-difference procedure. The Reynolds stress is modeled by an eddy viscosity function where the mixing length is assumed to depend on the distance from the wall and the boundary-layer thickness except in the near wall region where Van Driest's exponential damping function is used. For applications to wall blowing cases, comparisons with data show that the damping factor should be taken as a function of the blowing rate as determined from incompressible data. Nonequilibrium effects could be approximately accounted for by assuming that the mixing length in the midregion of the boundary layer is a function of a velocity profile shape factor. This shape factor is identical to the incompressible form factor, and the functional relation between it and the mixing length was based on incompressible data. The nearly quadratic relation between total temperature and velocity that is observed in wall boundary layers of hypersonic nozzles could be obtained in the calculations only by using values for the total turbulent Prandtl number of 1.5 or more. Tentative models for the fluctuating density terms were used in some of the calculations. The results indicated that these terms may have large effects on low-density, hypersonic boundary layers.

20. **Hixon, Barbara A.; Beckwith, Ivan E.; and Bushnell, Dennis M.: Computer Program for Compressible Laminar or Turbulent Non-Similar Boundary Layers. NASA TM X-2140, 1971.**

A computer program is described which solves the two-dimensional and axisymmetric forms of the compressible-boundary-layer equations for continuity, mean momentum, and mean total enthalpy by an implicit finite-difference procedure. Turbulent flow is treated by the inclusion of an eddy viscosity model based upon a mixing-length formulation. The eddy conductivity is related to the eddy viscosity by the turbulent Prandtl number which may be an arbitrary function of the distance from the wall. The laminar-boundary-layer equations are recovered when the eddy viscosity is zero. Since a finite-difference procedure is used, the effects of variable wall and edge boundary conditions are easily included by modifying the program inputs.

21. **Hunt, James L.; Bushnell, Dennis M.; and Beckwith, Ivan E.: Of The Compressible Turbulent Boundary Layer on a Blunt Swept Slab with and without Leading Edge Blowing. NASA TN D-6203, March 1971.**
22. **Harris, J. E.: Numerical Solution of the Equations for Compressible, Laminar, Transitional, and Turbulent Boundary Layers and Comparisons with Experimental Data. NASA TR R-368, August 1971.**

23. **Bushnell, D. M.: Calculation of Relaxing Turbulent Boundary Layers Downstream of Tangential Slot Injection. AIAA Journal of Spacecraft and Rockets, vol. 8, no. 5, May 1971, pp. 550-552.**

This note describes the application of a simple calculation method for compressible relaxing slot flows. Agreement with the data indicates that the simplified approach described here applies to the near slot as well as the far slot regions of the flow. This approach could be used, for example, in parametric studies to estimate the effects on the relaxation process of slot-to-edge total temperature ratio, slot height to boundary-layer height, and slot Mach number. The present method uses the turbulent boundary-layer computational procedure with the conservation equations for the mean flow solved by an implicit finite difference procedure and a turbulent Prandtl number relating the eddy viscosity and eddy conductivity. The solutions presented herein use a static turbulent Prandtl number of 0.9. Note that the present method applies only to the matched static pressure case where the exit pressure of the slot flow essentially equals the value in the external flow, i.e., where mixing dominates the problem.

24. **Bushnell, D. M.; and Alston, D. W.: Calculation of Compressible Adverse Pressure Gradient Turbulent Boundary Layers. Technical Note AIAA Journal, vol. 10, no. 2, February 1972, pp. 229-230.**

The present note reports on initial results of a study to determine if any of the possible reasons mentioned is the major cause of the disagreement between theory and data for the profile shapes in large adverse pressure gradient. The basic method has been modified to calculate the boundary layer in physical coordinates (x,y) rather than transformed variables. Also, the present program incorporates the capability to input pressure as a function of x and y . If one obtains the static pressure distribution either from data or approximate techniques, then the local density in the boundary layer is evaluated using the correct pressure level.

25. **Price, J. M.; and Harris, Julius E.: Computer Program for Compressible, Laminar, Transitional, or Turbulent Nonsimilar Boundary Layers. NASA TM X-2458, 1972.**

A computer program is described which solves the compressible laminar, transitional, or turbulent boundary-layer equations for planar or axisymmetric flows. Three-point implicit difference relations are used to reduce the momentum and energy equations to finite-difference form. These equations are solved simultaneously without iteration. Turbulent flow is treated by the inclusion of either a two-layer eddy-viscosity model or a mixing-length formulation. The eddy conductivity is related to the eddy viscosity through a static turbulent Prandtl number which may be an arbitrary function of the distance from the wall boundary. The transitional boundary layer is treated by the inclusion of an intermittency function which modifies the fully turbulent model. The laminar-boundary-layer equations are recovered when the intermittency is zero, and the fully turbulent equations are solved when the intermittency is unity. This program was used to obtain the solutions presented in NASA TR R-368.

26. **Bushnell, D. M.; and Alston, D. W.: Calculation of Transitional Boundary Layer Flows. Technical Note, AIAA Journal, vol. 1, no. 4, April 1973, pp. 554-556.**

27. **Fivel, H. J.; and Masek, R. V.: Analytical Comparison of Hypersonic Flight and Wind Tunnel Viscous/Inviscid Flow Fields. NASA CR-132359, July 1973.**

Flow fields were computed about blunted 0.525 and 0.697 radians (30 and 40 degree) cone configurations to assess the effects of nonequilibrium chemistry on the flow field geometry, boundary layer edge conditions, boundary layer profiles, and heat transfer and skin friction. Analysis were conducted at typical Space Shuttle entry conditions for both laminar and turbulent boundary layer flow. The viscous computer program used in this investigation was a modification of the Blottner non-similar viscous code which incorporated a turbulent eddy viscosity model after Cebeci. This report is a User's Manual for the viscous code and includes an analysis of the modifications to the original computer program, the input data setup instructions, a program listing, and flow chart. Also included are the results of a demonstration check-case for hydrogen-air combustion products along an exhaust nozzle.

28. **Cary, Aubrey M; and Bertram, Mitchel H.: An Evaluation of Engineering Methods for Predicting High-Speed Turbulent Skin Friction and Heat Transfer. NASA TN D-7507, 1974.**

A large collection of experimental turbulent-skin-friction and heat-transfer data for flat plates and cones was used to determine the most accurate of six of the most popular engineering-prediction methods; the data represent a Mach number range from 4 to 13 and ratio of wall to total temperature ranging from 0.1 to 0.7. The Spalding and Chi method incorporating virtual-origin concepts was found to be the best prediction method for Mach numbers less than 10; the limited experimental data for Mach numbers greater than 10 were not well predicted by any of the engineering methods except the Coles method.

29. **Fivel, H. J.; Masek, R. V.; and Mockapetris, L. J.: Analytical Comparison of Hypersonic Flight and Wind Tunnel Viscous/Inviscid Flow Fields. NASA CR-2489, February 1975.**

Flow fields were computed about blunted 0.524 and 0.698 radians (30 and 40 degree) cone configurations to assess the effects of nonequilibrium chemistry on the flow field geometry, boundary layer edge conditions, boundary layer profiles, and heat transfer and skin friction. Analysis were conducted at typical Space Shuttle entry conditions for both laminar and turbulent boundary layer flow. In these calculations, a wall temperature of 1365°F (2000°F) was assumed. The viscous computer program used in this investigation was a modification of the Blottner non-similar viscous code which incorporated a turbulent eddy viscosity model after Cebeci. The results were compared with equivalent calculations for similar (scaled) configurations at typical wind tunnel conditions. Wind tunnel test gases included air, nitrogen, CF₄, and helium. The viscous computer program used for wind tunnel conditions was the Cebeci turbulent non-similar computer code.

30. **Hunt, J. L.; and Souders, S. W.: Normal- and Oblique-Shock Flow Parameters in Equilibrium Air Including Attached-Shock Solutions for Surfaces at Angles of Attack, Sweep, and Dihedral. NASA SP-3093, 1975.**

Normal- and oblique-shock flow parameters for air in thermochemical equilibrium are tabulated as a function of shock angle for altitudes ranging from 15.24 km to 91.44 km in increments of 7.62 km at selected hypersonic speeds. Post-shock parameters tabulated include flow-deflection angle, velocity, Mach number, compressibility factor, isentropic exponent, viscosity, Reynolds number, entropy difference, and static pressure, temperature, density, and enthalpy ratios across the shock. A procedure is presented for obtaining oblique-shock flow properties in equilibrium air on surfaces at various angles of attack, sweep, and dihedral by use of the two-dimensional tabulations. Plots of the flow parameters against flow-deflection angle are presented at altitudes of 30.48, 60.96, and 91.44 km for various stream velocities.

31. **Harris, J. E.: An Implicit Finite-Difference Procedure for Solving the Three-Dimensional Compressible Laminar, Transitional, and Turbulent Boundary Layer Equations. Aerodynamic Analysis Requiring Advanced Computers, Part I, NASA SP-347, 1975, pp.19-40.**
32. **Bushnell, D. M.; Cary, A. M., Jr.; and Harris, J. E.: Calculation Methods for Compressible Turbulent Boundary Layers. Von Karman Institute for Fluid Dynamics (AGARD) Lecture Series Rhode-St. Genese, Belgium, March 1-5, 1976. NASA SP-422, 1977.**
33. **Watson, R. D.: Prediction of Outer Layer Mixing Lengths in Turbulent Boundary Layers. AIAA Journal, vol. 15, no. 4, April 1977, pp. 591-592.**

The purpose of this Note is to compare mixing length predictions by Pletcher's method with some recent values derived from turbulent boundary-layer profiles.

34. **Pittman, J. L.; and Dillon, J. L.: Vortex Lattice Prediction of Subsonic Aerodynamics of Hypersonic Vehicle Concepts. J. Aircraft, vol. 14, no. 10, October 1977, pp. 1017-1018.**

The vortex lattice method introduced by Lamar and Gloss (1975) was applied to the prediction of subsonic aerodynamic characteristics of hypersonic body-wing configurations. The reliability of the method was assessed through comparison of the calculated and observed aerodynamic performances of two National Hypersonic Flight Research Facility craft at Mach 0.2. The investigation indicated that a vortex lattice model involving 120 or more panel elements can give good results for the lift and induced drag coefficients of the craft, as well as for the pitching moment at angles of attack below 10 to 15 deg. Automated processes for calculating the local slopes of mean-chamber surfaces may also render the method suitable for use in preliminary design phases.

35. **Cary, A. M., Jr.; Bushnell, D. M.; and Hefner, J. N.: Calculation of Turbulent Boundary Layers With Tangential Slot Injection. ASME Paper 77-WA/HT-29, November 1977.**

Paper describes a numerical calculation scheme for tangential slot injection (wall-wake) flows; application of the scheme over a wide range of flow conditions indicates increased accuracy compared to previous work. Predictions from the numerical code were in good agreement with experiment (velocity profile, skin-friction, and effectiveness data) for low- and high-speed flows. To achieve improved accuracy, modifications in the turbulence modeling, compared to previous research, were necessary for the imbedded shear layer region in the near field and for the wall region near shear layer impingement. Anomalous behavior was noted for far field experimental velocity profiles in low-speed flow when the slot-to-free stream velocity ratio was near one.

36. **Stack, Sharon H.; Edwards, Clyde W.; and Small, William J.: GEMPAK: An Arbitrary Aircraft Geometry Generator. NASA TP-1022, December 1977.**

A computer program, "GEMPAK," has been developed to aid in the generation of detailed configuration geometry. The program was written to allow the user as much flexibility as possible in his choices of configurations and the detail of description desired and at the same time keep input requirements and program turnaround and cost to a minimum. The program consists of routines that generate fuselage and planar-surface (winglike) geometry and a routine that will determine the true intersection of all components with the fuselage. This paper describes the methods by which the various geometries are generated and provides input description with sample input and output. Also included are descriptions of the primary program variables and functions performed by the various routines. The FORTRAN program GEMPAK has been used extensively on the Control Data Corporation 6000 series computers in conjunction with interfaces to several aerodynamic and plotting computer programs and has proven to be an effective aid in the preliminary design phase of aircraft configurations.

37. **Edwards, C. L.; Meissner, F. T.; and Hall, J. B.: The Use of Computer-Generated Color Graphic Images for Transient Thermal Analysis. NASA TP-1455, July 1979.**

Color computer graphics techniques were investigated as a means of rapidly scanning and interpreting large sets of transient heating data. The data presented in this paper were generated to support the conceptual design of a heat-sink thermal protection system (TPS) for a hypersonic research airplane. Color-coded vector and raster displays of the numerical geometry used in the heating calculations were employed to analyze skin thicknesses and surface temperatures of the heat-sink TPS under a variety of trajectory flight profiles. Both vector and raster displays proved to be effective means for rapidly identifying heat-sink mass concentrations, regions of high heating, and potentially adverse thermal gradients. The color-coded (raster) surface displays are a very efficient means for displaying surface-temperature and heating histories, and thereby the more stringent design requirements can quickly be identified. The related hardware and software developments required to implement both the vector and the raster displays for this application are also discussed.

38. **Watson, R. D.: Generalized Velocities in the Outer Region of Hypersonic Turbulent Boundary Layers. AIAA Journal, vol. 17, no. 8, August 1979.**

An attempt is made to show that the outer portion of the velocity profile of hypersonic turbulent boundary layers can be transformed so that the constants determined by a best fit to the law of the wake are in reasonable agreement with the wake constant for incompressible boundary layers at the same Reynolds number. Both y transformations (where y is distance from the surface) and velocity transformations produce velocity profiles which, with the proper choice of wall shear stress to give shear velocity, can be reduced to the incompressible law of the wall.

39. **Anderson, M. S.; and Stroud, W. J.: A General Panel Sizing Computer Code and Its Application to Composite Structural Panels. AIAA Journal, vol. 17, no. 8, August 1979, pp. 892-897.**

A computer code for obtaining the dimensions of optimum (least mass) stiffened composite structural panels is described. The procedure, which is based on nonlinear mathematical programming and a rigorous buckling analysis, is applicable to general cross sections under general loading conditions causing buckling. A simplified method of accounting for bow-type imperfections is also included. Design studies in the form of structural efficiency charts for axial compression loading are made with the code for blade and hat stiffened panels. The effects on panel mass of imperfections, material strength limitations, and panel stiffness requirements are also examined. Comparisons with previously published experimental data show that accounting for imperfections improves correlation between theory and experiment.

40. **Pittman, Jimmy L.: The Application of Supersonic Linear Theory and Hypersonic Impact Methods to Three Nonslender Hypersonic Aircraft Concepts at Mach Numbers From 1.1 to 2.86. NASA TP-1539, December 1979.**

Aerodynamic predictions from supersonic linear theory and hypersonic impact theory were compared with experimental data for three hypersonic research airplane concepts over a Mach number range from 1.10 to 2.86. The linear theory gave good lift prediction and fair to good pitching-moment prediction over the Mach number (M) range. The tangent-cone theory predictions were good for lift and fair to good for pitching moment for $M = 2.0$. The combined tangent-cone/tangent-wedge method gave the least accurate prediction of lift and pitching moment. For all theories, the zero-lift drag was overestimated, especially for $M = 2.0$. The linear theory drag prediction was generally poor, with areas of good agreement only for $M = 1.2$. For $M = 2.0$, the tangent-cone method predicted the zero-lift drag most accurately.

41. **Edwards, C. L. W.: Version of Tangent Cone Approximation Developed at LaRC. (Appendix to NASA TP-1539), December 1979.**

42. **Zoby, E. V.; and Mass, J. N.: Thermodynamic Equilibrium Air Correlations for Flowfield Applications. AIAA Paper 81-0280, January 1981.**

Equilibrium-air thermodynamic correlations have been developed for flowfield calculation procedures. A comparison between the post shock results computed by the correlation equations and detailed chemistry calculations is very good. The thermodynamic correlations are incorporated in an approximate inviscid flowfield code with a convective heating capability for the purpose of defining the thermodynamic environment through the shock layer. Comparisons of heating rates computed by the approximate code and a viscous-shock-layer method are good. In addition to presenting the thermodynamic correlations, the impact of several viscosity models on the convective heat transfer is demonstrated.

43. **Stack, S. H.: A Computer-Aided Design System Geared Toward Conceptual Design in a Research Environment. AIAA Paper 81-0372, January 1981.**

44. **Zoby, E. V.; Moss, J. N.; and Sutton, K.: Approximate Convective Heating Equations for Hypersonic Flows. Journal of Spacecraft and Rockets, vol. 18, no. 1, January-February 1981, p. 64.**

Laminar and turbulent heating-rate equations appropriate for engineering predictions of the convective heating rates about blunt re-entry spacecraft at hypersonic conditions are developed. The approximate methods are applicable to both nonreacting and reacting gas mixtures for either constant- or variable-entropy edge conditions. A procedure which accounts for variable-entropy effects and is not based on mass balancing is presented. Results of the approximate heating methods are in good agreement with available experimental results as well as boundary-layer and viscous-shock-layer solutions.

45. **Drummond, J. P.; and Weidner, Elizabeth H.: Numerical Study of a Scramjet Engine Flow Field. AIAA Paper 81-0186, January 1981.**

A computer program has been developed to analyze the turbulent reacting flow field in a two-dimensional scramjet engine configuration. The program numerically solves the full two-dimensional Navier-Stokes and species equations in the engine inlet and combustor, allowing consideration of flow separation and possible inlet-combustor interactions. The current work represents an intermediate step towards development of a three-dimensional program to analyze actual scramjet engine flow fields. Results from the current program are presented that predict the flow field for two inlet-combustor configurations, and comparisons of the program with experiment are given to allow assessment of the modeling that is employed.

46. **Kumar, Ajay: Two-Dimensional Analysis of a Scramjet Inlet Flowfield. AIAA Paper No. 81-0185, January 1981.**

A computer code has been developed to analyze the flow in a supersonic combustion ramjet (scramjet) inlet using the two-dimensional Navier-Stokes equations. A numerical coordinate transformation is used to generate boundary-fitted curvilinear coordinates. Turbulence is modeled by a two-layer eddy viscosity model. The governing equations are solved by an explicit, finite-difference method. This code can analyze both inviscid and viscous flow with no strut, one strut, or multiple struts in the flowfield. The application of the two-dimensional analysis in the preliminary parametric design studies of the scramjet inlet is discussed briefly and results are presented for several inlet configurations.

47. **Moss, J. N.; and Kumar, A.: Significance of Turbulence and Transition Location on Radiative Heating with Ablation Injection. AIAA Paper No. 81-0281, January 1981.**

This paper presents forebody flowfield solutions for Jupiter entry conditions where the ablation injection rate is coupled with the surface heating rate. The calculations are made with a time-dependent viscous-shock-layer analysis where the flow is assumed to be in chemical equilibrium. Both laminar and turbulent solutions are presented to describe the impact of turbulence on surface mass loss rates for flow conditions where the heating is due primarily to radiation. Results are also presented where the transition location to turbulent flow is arbitrarily varied. Results show that the radiative heating rate values based on several downstream transition locations adjust quickly to the corresponding values based on transition near the stagnation point.

48. **Kaufman, L. G., II; and Johnson, C. B.: Methods for Estimating Pressure and Thermal Loads Induced by Elevon Deflections on Hypersonic Vehicle Surfaces With Turbulent Boundary Layers. NASA TM-83130, September 1981.**

Empirical analytic methods are presented for calculating thermal and pressure distributions in three-dimensional, shock-wave turbulent-boundary-layer, interaction-flow regions on the surface of controllable hypersonic aircraft and missiles. The methods, based on several experimental investigations, are useful and reliable for estimating both the extent and magnitude of the increased thermal and pressure loads on the vehicle surfaces.

49. **Kumar, Ajay: Numerical Analysis of the Scramjet Inlet Flow Field by Using Two-Dimensional Navier-Stokes Equations. NASA TP-1940, December 1981.**

A computer code has been developed to solve the full two-dimensional Navier-Stokes equations in a supersonic combustion ramjet (scramjet) inlet. In order to be able to consider a general inlet geometry with embedded bodies, a numerical coordinate transformation is used which generates a set of boundary-fitted curvilinear coordinates. The explicit finite-difference algorithm of MacCormack is used to solve the governing equations. An algebraic, two-layer eddy-viscosity model is used for the turbulent flow. The code can analyze both inviscid and viscous flows with no strut, one strut, or multiple struts in the flow field. The application of the two-dimensional analysis in the preliminary parametric design studies of a scramjet inlet is discussed briefly. Detailed results are presented for one model problem and for several actual scramjet-inlet configurations.

50. **Kumar, Ajay: Three-Dimensional Inviscid Analysis of the Scramjet Inlet Flow Field. AIAA Paper 82-0060, January 1982.**

A computer code has been developed to analyze the inviscid flow field in a supersonic combustion ramjet (scramjet) inlet. The code uses the three-dimensional Euler equations in full conservation form to describe the inlet flow. An algebraic numerical coordinate transformation is used to generate a set of boundary-fitted curvilinear coordinates. The governing equations are solved by a time-asymptotic, unsplit, two-step, finite-difference method. This method is highly efficient on the vector processing computers for which the current code is written. Detailed results are presented for two scramjet inlet configurations over a range of Mach numbers. The calculated results are compared with the available experimental and theoretical results.

51. **Kumar, Ajay: Two-Dimensional Analysis of the Inlet Flow of a Scramjet Engine. AIAA Journal, vol. 20, January 1982, pp. 96-97.**

A computer code has been developed to analyze the flow in a supersonic combustion ramjet (scramjet) inlet using the two-dimensional Navier-Stokes equations. A numerical coordinate transformation is used to generate boundary-fitted curvilinear coordinates. Turbulence is modeled by a two-layer eddy viscosity model. The governing equations are solved by an explicit, finite-difference method. This code can analyze both inviscid and viscous flow with no strut, one strut, or multiple struts in the flowfield. The application of the two-dimensional analysis in the preliminary parametric design studies of the scramjet inlet is discussed briefly and results are presented for several inlet configurations.

52. **Harris, J. E.; and Blanchard, D. K.: Computer Program for Solving Laminar, Transitional, or Turbulent Compressible Boundary-Layer Equations for Two-Dimensional and Axisymmetric Flow. NASA TM-83207, February 1982.**

A computer program is described which solves the compressible laminar, transitional, or turbulent boundary-layer equations for two-dimensional or axisymmetric perfect gas flows. Three-point implicit difference relations are used to reduce the momentum and energy equations to finite-difference form for simultaneous solution. The software, program VGBLP, is a modification of the approach presented in NASA TR R-368 and NASA TM X-2458, respectively. Turbulence-closure options include either two-layer eddy-viscosity or mixing-length models. Eddy conductivity is modeled as a function of eddy viscosity through a static turbulent Prandtl number formulation. Several options are provided for specifying the static turbulent Prandtl number. The transitional boundary layer is treated through a streamwise intermittency function which modifies the turbulence-closure model. This model is based on the probability distribution of turbulent spots and ranges from zero to unity for laminar and turbulent flow, respectively. Several test cases are presented as guides for potential users of the software.

53. **Kumar, Ajay: Three-Dimensional Inviscid Analysis of the Scramjet Inlet Flow Field. Journal of Aircraft, vol. 19, October 1982, pp. 892-893.**

A comprehensive research program is underway at the NASA Langley Research Center to develop an airframe-integrated, hydrogen-fueled supersonic combustion ramjet (scramjet) engine for hypersonic propulsion. The current scramjet concept uses a rectangular module approach which has a fixed-geometry inlet with swept, wedge-shaped sidewalls. The sweep of the sidewalls, in combination with a recess in the cowl, allows some flow to spill out ahead of the the cowl, thus producing a variable-geometry inlet. This provides the potential to operate over a range of Mach numbers even with the fixed-geometry inlet.

54. **Moss, J. N.; and Kumar, Ajay: Significance of Turbulence and Transition Location on Radiative Heating with Ablation Injection. AIAA Progress in Astronautics and Aeronautics, vol. 82, 1982, pp. 354-381.**

This paper presents forebody flowfield solutions for Jupiter entry conditions where the ablation injection rate is coupled with the surface heating rate. The calculations are made with a time-dependent viscous-shock-layer analysis where the flow is assumed to be in chemical equilibrium. Both laminar and turbulent solutions are presented to describe the impact of turbulence on surface mass loss-rates for flow conditions where the heating is due primarily to radiation. Results are also presented where the transition location to turbulent flow is arbitrarily varied. Results show that the radiative heating rate values based on several downstream transition locations adjust quickly to the corresponding values based on transition near the stagnation point.

55. **Kumar, Ajay: Numerical Analysis of Scramjet Inlet Flow Field Using the Three Dimensional Navier Stokes Equations. Presented at the 1983 JANNAF Propulsion Meeting, Monterey, California, February 14-18, 1983.**

A computer program has been developed to analyze supersonic combustion ramjet (scramjet) inlet flow fields. The program solves the three-dimensional Euler or Navier-Stokes equations in full conservation form by a well-known explicit, predictor-corrector technique. Turbulence is modeled by an algebraic eddy-viscosity model. Detailed laminar and turbulent flow results are presented for a symmetric wedge corner and a comparison is made with the available experimental results to allow assessment of the program. Results are then presented for an actual scramjet inlet configuration.

56. **Clever, W. C.; and Shankar, V.: Nonlinear Potential Analysis Techniques for Supersonic/Hypersonic Configuration Design. NASA CR-1656078, March 1983.**

Approximate nonlinear inviscid theoretical techniques for predicting aerodynamic characteristics and surface pressures for relatively slender vehicles at moderate hypersonic speeds were developed. Emphasis was placed on approaches that would be responsive to preliminary configuration design level of effort. Second order small disturbance and full potential theory was utilized to meet this objective.

57. **Edwards, Clyde L. W.; and Cole, Stanley R.: Entry Heating Predictions for Shuttle Orbiter Lower Surface. NASA TM-84624, July 1983.**

A broad base of thermocouple and phase change paint data was assembled and correlated to the nominal design 14414.1 and proposed STS-1 (first flight of the space transportation system) entry trajectories. Averaged data from phase change paint tests compared favorably with thermocouple data for predicting heating rates. Laminar and turbulent radiation equilibrium heating rates were computed on the lower surface of the Shuttle orbiter for both trajectories, and the lower surface center line results were compared both with aerodynamic heating design data and with flight values from the STS-1 and STS-2 trajectories. The peak laminar heating values from the aerodynamic heating design data book were generally 40 to 60 percent higher than the laminar estimates of this study, except at the 55 percent location of maximum span where the design data book values were less than 10 percent higher. Estimates of both laminar and turbulent heating rates compared favorably with flight data.

58. **Rathjen, K. A.; and Burk, H. O.: Enhancement of the CAVE Computer Code. NASA CR-3731, December 1983.**

The computer program CAVE (Conduction Analysis via Eigenvalues) is a convenient and efficient computer code for predicting two-dimensional temperature histories within thermal protection systems for hypersonic vehicles. The capabilities of CAVE were enhanced by incorporation of the following features into the code: real-gas effects in the aerodynamic heating predictions, geometry and aerodynamic heating package for analysis of cone-shaped bodies input option to change from laminar to turbulent heating predictions on leading edges, modification to account for reduction in adiabatic wall temperature with increase in leading sweep, geometry packages for two dimensional scramjet engine sidewall, with an option for heat transfer to external and internal surfaces, printout modification to provide tables of select temperatures for plotting and storage, and modifications to the radiation calculation procedure to eliminate temperature oscillations induced by high heating rates. These new features are described.

59. **Zoby, E. V.; and Simmonds, A. L.: Engineering Flowfield Method With Angle-of-Attack Applications. AIAA Paper 84-0393, January 1984.**

An approximate inviscid flowfield method has been extended to include heat-transfer predictions and a technique to account for the effect of variable-entropy edge conditions on the heat transfer. Results of the approximate code have been validated by comparison with experimental data and results of detailed predictions. The engineering code computes the inviscid flowfield and convective heating over hyperboloids, ellipsoids, paraboloids, and sphere-cones at zero deg. angle of attack. An application to angle-of-attack conditions is included in the present method by using existing approximations to: (1) account for the streamline-spreading effects on the heat transfer along the windward and leeward rays of sphere-cones and (2) to compute the corresponding circumferential heating. Present results of the engineering calculations are shown to be in good agreement with existing pressure and heating data over sphere-cones, even at high incidence values, with the restriction that the sonic-line location remain on the spherical cap. The present technique has been demonstrated to provide a rapid but reliable method for predicting surface-measurable quantities and flow properties through the shock layer. The code represents a versatile engineering method for parametric or preliminary thermal design studies.

60. **Kumar, Ajay; and Trexler, C. A.: Analysis and Projection of the Performance of Scramjet Inlets Utilizing a Three-Dimensional Navier-Stokes Code. Presented at 1984 JANNAF Propulsion Meeting, February 1984.**

A series of inlet analysis codes (2-D, axisymmetric, 3-D) were developed which can analyze complicated flow through complex inlet geometries in reasonably efficient manner. The codes were verified and are being used extensively to analyze practical inlet geometries. Newly installed VPS 32 computers allow more complex configurations to be analyzed. Scalar FORTRAN versions are available to increase transportability of the codes for use on other Scalar computers and on Gray vector processing computer.

61. **Kumar, Ajay: User's Guide for NASCRIN - A Vectorized Code for Calculating Two-Dimensional Supersonic Internal Flow Fields. NASA TM-85708, February 1984.**

A computer program NASCRIN has been developed for analyzing two-dimensional flow fields in high-speed inlets. It solves the two-dimensional Euler and Navier-Stokes equations in conservation form by an explicit, two-step finite-difference method. An explicit-implicit method can also be used at the user's discretion for viscous flow calculations. For turbulent flow, an algebraic, two-layer eddy-viscosity model is used. The code is operational on the CDC® CYBER 203 computer system and is highly vectorized to take full advantage of the vector-processing capability of the system. It is highly user oriented and is structured in such a way that for most supersonic flow problems, the user has to make only a few changes. Although the code is primarily written for supersonic internal flow, it can be used with suitable changes in the boundary conditions for a variety of other problems.

62. **Shankar, V.; and Clever, W. C.: Nonlinear Potential Analysis Techniques for Supersonic/Hypersonic Aerodynamic Design. NASA CR 172299, March 1984.**

Approximate nonlinear inviscid theoretical techniques for predicting aerodynamic characteristics and surface pressures for relatively slender vehicles at subsonic moderate hypersonic speeds were developed. Emphasis was placed on approaches that would be responsive to conceptual configuration design level of effort. Second order small disturbance and full potential theory was utilized to meet this objective.

63. **Talcott, Noel A., Jr.; and Kumar, Ajay: Numerical Simulation of Flow Through Inlets/Diffusers with Terminal Shocks. AIAA Paper 84-1362, June 1984.**

Two-dimensional Euler and Navier-Stokes solutions of the flow through three inlet/diffuser configurations with terminal shock systems are reported. Calculations without bleed indicate that the terminal shock location is very sensitive to the outflow back pressure. For cases where there are no available experimental results, it becomes difficult to estimate the back pressure that will result in a terminal shock. Estimates based on quasi-one-dimensional analysis are not found adequate for complex two-dimensional flows. It is found that since the flow downstream of the terminal shock is subsonic, and what happens at the outflow boundary affects the flow inside the inlet, enough of the subsonic diffuser must be modeled to accurately predict the terminal shock region. The diffuser portion should be fairly long with the outflow boundary occurring in a region of more or less uniform flow to be able to prescribe a uniform back pressure. The third configuration studied was investigated with and without incorporating bleed in the code. It is found that the use of bleed stabilizes the shock location and allows solutions which without bleed result in unstarting of the inlet. Comparisons are made with available experimental data.

64. Moss, J. N.; and Bird, G. A.: **Direct Simulation of Transitional Flow for Hypersonic Reentry Conditions.** AIAA Progress in Aeronautics and Astronautics Series, vol. 96, 1985, pp. 113-139.

This paper presents results of flowfield calculations for typical hypersonic reentry conditions encountered by the nose region of the Space Shuttle Orbiter. Most of the transitional flow regime is covered by the altitude range of 150 to 92 km. Calculations were made with the Direct Simulation Monte Carlo (DSMC) method that accounts for translational, rotational, vibrational, and chemical nonequilibrium effects. Comparison of the DSMC heating results with both Shuttle flight data and continuum predictions showed good agreement at the lowest altitude considered. However, as the altitude increased, the continuum predictions which did not include slip effects, departed rapidly from the DSMC results by overpredicting both heating and drag. The results demonstrate the effects of rarefaction on the shock and the shock layer, along with the extent of the slip and temperature jump at the surface. Also, the sensitivity of the flow structure to the gas-surface interaction model, thermal accommodation, and surface catalysis are studied.

65. Gupta, R. N.; Scott, D. C.; and Moss, J. N.: **Surface-Slip Equations for Low Reynolds Number Multicomponent Air Flows.** AIAA Progress in Astronautics and Aeronautics Series, vol. 96, 1985, pp. 465-490.

The surface-slip equations have been obtained from the closed form solutions of the mass, momentum, and energy flux equations using the Chapman-Enskog velocity distribution function. This function represents a solution of the Boltzmann equation in the Navier-Stokes approximation. The obtained expressions provide jump (or slip) in the wall values of species concentration, pressure, velocity, and temperature for the low Reynolds number high-altitude regime of a space vehicle. The analysis includes multicomponent diffusion with finite-rate surface catalytic recombination. A consistent set of equations is provided for multicomponent, binary, and single-species mixtures. Expression is also provided for finite-rate surface catalytic recombination without slip for a multicomponent mixture.

66. **DeJarnette, F. R.; Hamilton, H. H.; Weilmuenster, K. J.; and Cheatwood, F. M.: A Review of Some Approximate Methods Used in Aerodynamic Heating Analysis. AIAA Paper 85-0906, June 1985.**

Rapid, approximate methods for calculating laminar, transitional, and turbulent heating rates on three-dimensional vehicles are discussed. For the windward plane of symmetry, several methods give reasonably accurate heating rates. Surface pressures can be calculated from modified Newtonian theory, the tangent-cone method, and other methods using the equivalent asymmetric body concept. For regions off the windward symmetry plane, accurate, inviscid, finite-difference methods like the HALIS code are needed for large angles of attack. Three-dimensional boundary-layer effects may be accounted for by using the axisymmetric analogue. Laminar heating rates are predicted well by the local similarity methods of Lees or Beckwith and Cohen. Turbulent heating rates calculated by the Van Driest II or reference enthalpy method of Zoby, Moss, and Sutton compare well with experimental and shuttle flight data. Variable entropy at the edge of the boundary layer can be determined by a mass balance technique or interpolating in the inviscid flow field at a distance equal to the boundary-layer thickness from the wall. Approximate methods are still needed to calculate heating rates for low Reynolds number flows, nonequilibrium chemistry, and separated flow regions.

67. **Kumar, A.: Numerical Simulation of Flow Through Scramjet Inlets Using a Three-Dimensional Navier-Stokes Code. AIAA Paper No. 85-1664, July 1985.**

A three-dimensional Reynolds-averaged Navier-Stokes code has been used to numerically analyze flow through a two-strut, supersonic combustion ramjet (scramjet) inlet configuration. It solves the governing equations in full conservation form using either the fully explicit or explicit-implicit method due to MacCormack. An algebraic two-layer eddy-viscosity model is used for turbulent flow calculations. The analysis allows inclusion of end effects which are caused by the aft placement of the cowl on the underside of the inlet. A special grid has been developed to accommodate the struts embedded in the inlet flow field. Detailed numerical results are presented here for the two-strut configuration, and a comparison is made with the available experimental results.

68. **Talcott, N. A.; and Kumar, Ajay: Two-Dimensional Viscous Simulation of Inlet/Diffuser Flows With Terminal Shock. Journal of Propulsion and Power, vol. 1, no. 2, March-April 1985, pp. 103-108.**

Two-dimensional Navier-Stokes solutions of the flow through three inlet/diffuser configurations with terminal shock systems are reported. Calculations without bleed indicate that the terminal shock location is very sensitive to the outflow back pressure. For cases where there are little or no available experimental results, it becomes difficult to estimate the back pressure that will result in a terminal shock. Estimates based on quasi-one-dimensional analysis are not found adequate for complex two-dimensional flows. It is found that since the flow downstream of the terminal shock is subsonic, and what happens at the outflow boundary affects the flow inside the inlet, enough of the subsonic diffuser must be modeled to accurately predict the terminal shock region. The diffuser portion should be fairly long with the outflow boundary occurring in a region of more or less uniform flow to be able to prescribe a uniform back pressure. The second configuration studied was investigated with and without incorporating bleed in the code. It is found that the use of bleed stabilizes the shock location and allows solutions which, without bleed result in unstarting of the inlet. The third configuration required a significant amount of bleed through the ramp and cowl surfaces (both ahead and behind the throat) to avoid separation and provide uniform flow at the engine-face station. Comparisons are made with available experimental data.

69. **Kumar, Ajay: Numerical Simulation of Scramjet Inlet Flow Fields. NASA TP-2517, May 1986.**

A computer program has been developed to analyze supersonic combustion ramjet (scramjet) inlet flow fields. The program solves the three-dimensional Euler or Reynolds averaged Navier-Stokes equations in full conservation form by either the fully explicit or explicit-implicit, predictor-corrector method of MacCormack. Turbulence is modeled by an algebraic eddy-viscosity model. The analysis allows inclusion of and effects which can significantly affect the inlet flow field. Detailed laminar and turbulent flow results are presented for a symmetric-wedge corner, and comparisons are made with the available experimental results to allow assessment of the program. Results are then presented for two inlet configurations for which experimental results exist at the NASA Langley Research Center.

70. **Kumar, Ajay; and Anderson, G. Y.: Study of Hypersonic Inlet Flow Fields With a Three-Dimensional Navier-Stokes Code. AIAA Paper 86-1426, June 1986.**

A three-dimensional Navier-Stokes code is used to study the flow through an inlet with a rectangular capture area, but with circular or essentially circular internal cross sections. The inlet may have a center body extending partially or fully over the length of the inlet. Calculations are made under a variety of geometrical constraints imposed on the inlet cross sections to study the flow quality in the inlet. It is observed that for this class of inlets, the boundary layer has a tendency to thicken and separate readily due to the adverse pressure gradient caused by the continuous compression in the inlet. Since no experimental results are yet available, this paper has presented only the numerical results from the present analysis.

SUBJECT CATEGORY 8 - TEST TECHNIQUES

1. **McLellan, C. H.; and Williams, T. W.: Liquefaction of Air in the Langley 11-Inch Hypersonic Tunnel. NACA TN 3302, October 1954.**
2. **Jones, R. A.; and Hunt, J. L.: Use of Temperature Sensitive Coatings for Obtaining Quantitative Aerodynamic Heat Transfer Data. AIAA Journal, vol. 2, July 1964, pp. 1354-1356.**
3. **Johnston, Patrick J.; and Witcofski, Robert D.: Effect of a Variable-Geometry Diffuser on the Operating Characteristics of a Helium Tunnel Designed for a Mach Number in Excess of 20. NASA TN D-237, February 1960.**
4. **Miller, Charles G. III: An Experimental Investigation of Support Interferences on a Blunt Body of Revolution at a Mach Number of Approximately 20. NASA TN D-2742, April 1965.**
5. **Jones, R. A.; and Hunt, J. L.: An Improved Technique for Obtaining Quantitative Aerodynamic Heat Transfer Data with Surface Coating Materials. Journal of Spacecraft and Rockets, vol. 2, no. 4, July 1965, pp. 632-634.**

A method for obtaining quantitative aerodynamic heat-transfer data by the use of a visible phase-change coating is described briefly. The coating materials undergo a phase change from an opaque solid to a clear liquid at known temperatures with an accuracy of ± 1 percent which is independent of both ambient pressure and heating rate. This note further discusses the accuracy of this method and presents additional experimental results.

6. **Jones, R. A.; and Hunt, J. L.: Use of Fusible Temperature Indicators for Obtaining Quantitative Aerodynamic Heat Transfer Data. NASA TR R-230, February 1966.**
7. **Hunt, J. L.; and Jones, R. A.: Comment on Measurement of Aerodynamic Heating of Wind Tunnel Models by Means of Temperature Sensitive Paint. Journal of Spacecraft and Rockets, vol. 8, no. 5, May 1966, pp. 767-768.**

In a previous note, a method for obtaining quantitative aerodynamic heat-transfer data on complicated models was described. A temperature-sensitive paint, which undergoes color changes at certain temperatures, was used in conjunction with a reference sphere for obtaining quantitative aerodynamic heat-transfer data on an X-20 glider model. However, no mention was made of the dependence of the color-change temperature on ambient pressure, which could introduce an error into the reference body method unless the pressure on the sphere and the test model were approximately equal at each location where a corresponding color change occurred.

8. **Sterrett, J. R.; Morrisette, E. Leon; Whitehead, A. H., Jr.; and Hicks, R.: Transition Fixing for Hypersonic Flow. NASA TN D-4129, October 1967.**

Present wind-tunnel facilities lack the capability to duplicate the Reynolds number associated with the hypersonic-cruise vehicle. As a means of overcoming this problem, attention is being given to artificial promotion of transition by means of surface roughness. At lower speeds, boundary-layer roughness has been used successfully. However, at hypersonic speeds, the required roughness height is so large that the method raises many questions. This paper considers these questions and examines the overall problems associated with boundary-layer "trips" employed to produce turbulence at hypersonic conditions.

9. **Harvey, William D.; and Bushnell, Dennis M.: Comments on Hypersonic Turbulent Boundary-Layer Measurements Using an Electron Beam. NASA SP-216, Symposium on Compressible Turbulent Boundary Layers, Paper No. 9, December 1968.**

Estimates of the root-mean-square velocity fluctuation level and trend at Mach 8.5 were made by utilizing the experimental fluctuating density and pitot pressure measurements of Wallace. Also, fair estimates of both the magnitude and trend of the fluctuating density and velocity can be made by using a mixing length type of approach. Present indications are that for zero pressure gradient the intensity of the fluctuating velocity in a boundary layer may be independent.

10. **Morrisette, E. Leon; Stone, David R.; and Cary, Aubrey M., Jr.: Downstream Effects of Boundary-Layer Trips in Hypersonic Flow. NASA SP-126, Symposium on Compressible Turbulent Boundary Layers, Paper No. 16, December 1968, pp. 437-453.**

Flow-field and boundary-layer surveys (pitot and total temperature) are shown for flow downstream of spherical roughness trips on a flat plate at near-adiabatic wall conditions in Mach 8.5 flow. The trips are shown to cause large distortions of the flow field and outer portion of the boundary layer. A method for minimizing these distortions is suggested. The nature of the tripping mechanism is discussed. Spherical and air jet trips are compared, and surface heating downstream of both types is presented.

11. **Bushnell, D. M.; and Harvey W. D.: Comments On Hypersonic Turbulent Boundary Layer Measurements Using An Electron Beam. NASA SP-216, Compressible Turbulent Boundary Layers, 1969, pp. 309-317.**

12. **Whitehead, Allen H., Jr.: Flow Field and Drag Characteristics of Several Boundary Layer Tripping Elements in Hypersonic Flow. NASA TN D-5454, October 1969.**

13. **Weinstein, L. M.; Wagner, R. D., Henderson, A. Jr.,; and Ocheltree, S. L.: Electron Beam Flow Visualization in Hypersonic Helium Flow. ICIASF '69, Proceedings, 3rd, May 5-8, 1969, pp. 72-78.**

14. **Jones, Robert A.; and Hunt, James L.: Use of Tetrafluoromethane to Simulate Real Gas Effects on the Hypersonic Aerodynamics of Blunt Vehicles. NASA TR R-312, June 1969.**

An experimental and theoretical study was made of the use of tetrafluoromethane (CF_4) in low hypersonic flows as a means of simulating the flow field over the forebody of blunt configurations in hypersonic flight where dissociation of the gas in the shock layer gives large density ratios across the shock wave. A pilot model hypersonic CF_4 blowdown tunnel at the Langley Research Center was built and studied. With this facility a normal-shock density ratio of 12.1 was obtained at a Mach number of 6 with stagnation conditions of 1650 N/cm^2 and 736° K . The results of the investigation indicate that flow fields over the forebody of blunt configurations for real-gas conditions resulting in normal-shock density ratios up to 17 can be simulated by this method. A computer program for isentropic flow of CF_4 as well as flow across normal and oblique shocks was set up. Plots of various flow parameters are presented as a function of free-stream Mach number for various stagnation temperatures. A method for computing axisymmetric nozzle contours which includes a correction for boundary-layer displacement thickness is described.

15. **Hunt, James L.; and Smith, K. A.: Calculation Procedure for Evaluating Isentropic Expansions and Flow Across Normal and Oblique Shocks in Tetrafluoromethane. Appendix A of NASA TR R-312, June 1969.**
16. **Hunt, James L.; and Boney, L. R.: Hypersonic Nozzle Contours for Tetrafluoromethane. Appendix B of NASA TR R-312, June 1969.**
17. **Dillon, James L.; Wright, R. L.; and Smith, G. L.: Data Reduction Technique for Determining Rapidly Rising Heating Rates to Thermally Thick Walls. NASA TN D-5309, July 1969.**
18. **Morrisette, E. L.; Stone, D. R.; and Whitehead, A. H., Jr.: Boundary Layer Tripping with Emphasis on Hypersonic Flow, Viscous Drag Reduction. C. Sinclair Wells, E., Plenum Press, 1969.**
19. **Beckwith, Ivan E.; Bushnell, Dennis M.; and Harvey William D.: Fluctuating Properties of Turbulent Boundary Layers for Mach Numbers Up To 9. NASA TN D-5496, October 5, 1969.**
20. **Whitehead, Allen H., Jr.: Flow-Field and Drag Characteristics of Several Boundary-Layer Tripping Elements in Hypersonic Flow.**

Flow-field and drag characteristics of several tripping-element shapes have been examined under laminar-boundary-layer conditions at a free-stream Mach number of 6.8 to assess the contribution of the elements to the drag of wind-tunnel models and to examine flow-field characteristics around individual element shapes. The element drag coefficient based on local condition depends primarily on element shape, the spacing between the elements, and the ratio of the element height to boundary-layer height. Advantageously, certain element shapes have drag coefficients which are relatively independent of the ratio of element height to boundary-layer height for values of this ratio that have practical application for hypersonic wind-tunnel facilities.

21. **Watson, Ralph D.; and Bushnell, Dennis M.: Calibration of the Langley Mach 20 High Reynolds Number Helium Tunnel Including Diffuser Measurements. NASA TM X-2353, 1971.**
22. **Hunt, J. L.; and Pitts, J. I.: Thin Wing Corrections for Phase Change Heat-Transfer Data. Journal of Spacecraft and Rockets, vol. 8, no. 12, December 1971.**

Since no methods are available for determining the magnitude of the errors incurred when the semi-infinite slab assumption is violated, a computer program was developed to calculate the heat-transfer coefficients to both sides of a finite, one-dimensional slab subject to the boundary conditions ascribed to the phase-change coating technique. The results have been correlated in the form of correction factors to the semi-infinite slab solutions in terms of parameters normally used with the technique. These correlations are not restricted to slab thickness or thermophysical properties and can be easily used to obtain accurate data on thin model sections.

23. **Creel, T. R., Jr.; and Hunt, J. L.: Photographing Flow Fields and Heat Transfer Patterns in Color Simultaneously. Astronautics and Aeronautics, vol. 10, April 1972, pp. 54-55.**

With the development of color schlieren, there has been developed at the Langley Research Center an experimental technique for obtaining aerodynamic heat-transfer data on a model. This technique, referred to as the phase-change-coating method, employs a temperature-sensitive material as a thin surface coating that undergoes a visible phase change from opaque to clear liquid at a known temperature. A schlieren system, either black and white or color, shows only the outline of the model. Conversely, the phase-change-coating technique using motion picture photography records model features but not shock shapes. An existing schlieren system was modified by the addition of a white light, a 22.5-deg. prism, two acromatic lenses, and a twin knife-edge (slit), to form a color schlieren. A high-intensity light, set up on the test-section side adjacent to the recording camera cast reflections from the model to the camera, thus showing details of the model.

24. **Woods, W. C.; and Arrington J. P.: Electron Beam Flow Visualization-Applications in the Definition of Configuration Aerothermal Characteristics. AIAA Paper 72-1016, September 1972.**

Comparisons between flow visualization systems using electron-beam fluorescence, schlieren, and shadowgraph techniques illustrate the advantages associated with the electron beam. Specific applications of this method as an aid in analyzing unusual force data and in defining origin of erosion on a heat-transfer model are cited. Results of combined electron-beam oil flow studies on configurations illustrate that the simultaneous definition of the external flow field and its surface flow can be obtained. Comparisons between the electron-beam oil-flow visualization method and phase-change coating heat-transfer tests on a shuttle ascent configuration indicate the complementary nature of these two testing techniques. Potential methods for improving the electron-beam technique are included.

25. **Stone, D. R.; Harris, J. E.; Throckmorton, D. A.; and Helms, V. T. III: Factors Affecting Phase-Change Paint Heat-Transfer Data Reduction With Emphasis on Wall Temperatures Approaching Adiabatic Conditions. AIAA Paper 72-1030, September 1972.**

Accurate knowledge of the adiabatic wall temperature is known to be important in the reduction of phase-change heat-transfer data. The assumption of an adiabatic wall temperature equal to the total temperature in this procedure results in low values of heat-transfer coefficients. As a basis for study of these effects, locally similar and nonsimilar numerical solutions of the compressible laminar boundary-layer equations were used to examine possible variations of recovery factor with test gas, local Mach number, and pressure gradient. The level and distribution of wall temperature are shown to have a significant effect on the convective heat transfer to models with large heat-transfer gradients. The analytical results are substantiated by an analysis of experimental data obtained on a hemisphere-cylinder and a representative Space Shuttle Orbiter at Mach 20 in helium.

26. **Hunt, J. L.; Pitts, J. I.; and Richie, C. B.: Application of Phase-Change Technique to Thin Sections with Heating on Both Surfaces. NASA TN D-7193, August 1973.**

A numerical digital computer program was developed to calculate the heat-transfer coefficients for both sides of a one-dimensional finite slab subject to the boundary conditions ascribed to the phase-change coating heat-transfer technique. In a typical tunnel test situation where a thin wing was exposed to heating on both sides, the data reduction procedures for a semi-infinite slab gave heat-transfer coefficients as much as 375 percent too high on the side with the lowest heating. The results from the one-dimensional finite-slab procedure are presented in the form of correction factors to the solution for a semi-infinite slab in terms of parameters normally used with the phase-change heat-transfer technique. These correlations are not restricted to slab thickness or thermophysical properties and are easily used to obtain accurate data on thin model sections.

27. **Jones, Robert A.; and Hunt, James L.: Measured Pressure Distributions on Large Angle Cones in Tetrafluoromethane, Air, and Helium at Hypersonic Speeds. NASA TN D-7429, December 1973.**

An experimental study of surface pressure distributions on a family of blunt and sharp large angle cones was made in hypersonic flows of helium, air, and tetrafluoromethane. The effective isentropic exponents of these flows were 1.67, 1.40, and 1.12. Thus, the effect of large shock density ratios such as might be encountered during planetary entry because of "real-gas" effects could be studied by comparing results in tetrafluoromethane with those in air and helium. It was found that shock density ratio had a large effect on both shock shape and pressure distribution. The differences in pressure distribution indicate that for atmospheric flight at high speed where "real-gas" effects produce large shock density ratios, large-angle cone vehicles can be expected to experience different trim angles of attack, drag coefficient, and lift-drag ratios than those for ground tests in air wind tunnels. Comparison of the data with several theories indicated that (1) for sharp cones having attached shock waves, the sharp-cone solutions provide a good prediction of pressure, and (2) for both sharp and blunt cones having subsonic flow over the forebody, the semiempirical, \sin^2 -deficiency method of Love gave the best prediction of pressure distribution.

28. **Wagner, Richard D.; and Weinstein, Leonard M.: Hot Wire Anemometry in Hypersonic Helium Flows. NASA TN D-7465, 1974.**

Hot-wire anemometry techniques are described that have been developed and used for hypersonic-helium-flow studies at the Langley Research Center. The short run time available dictated certain innovations in applying conventional hot-wire techniques. Some examples are given to show the application of the techniques used. Some of the modifications to conventional equipment are described, including probe modifications and probe heating controls.

29. **Hunt, James L.; Jones, Robert A.; and Midden, R. E.: Simulation of Real-Gas Effects for Mars Entry. Journal of Spacecraft and Rockets, vol. 11, no. 1, January 1974, pp. 62-64.**

The changes in aerodynamic characteristics due to real-gas effects associated with high speed flight (characterized by large shock density ratios) are primarily the result of changes in surface pressures acting on the forebody. The surface pressures are affected by a change in shock density ratio (real-gas effects) in two ways. First, the level of pressure at the stagnation point relative to freestream dynamic pressure is changed, and second, the distribution of surface pressure relative to stagnation-point pressure is changed. The density-ratio effect on the stagnation-point pressure level can be estimated by considering the flow of a perfect gas about a blunt body.

30. Hillard, M. E., Jr.; Morrisette, E. L.; and Emory, M. L.: Raman Scattering Applied to Hypersonic Air Flow. *AIAA Journal*, vol. 12, no. 8, August 1974, pp. 1160-1163.

In the present investigation, the Raman scattering technique was used to measure the local static temperature and the gas number density over a flat plate in a Mach 5 nozzle of the Langley nozzle test chamber with air as the test gas. While this flowfield is not three-dimensional, the accuracy of the resulting measurements of density and temperature confirm the accuracy of the technique for more complicated flows (as long as the spatial resolution of the sample volume is sufficiently small). Measurements were made in the inviscid flowfield of a sharp leading-edge flat-plate model at several angles of attack (-5° to 15°) and over a wide range of tunnel conditions (stagnation pressures P_0 from 1.7×10^5 to 2.8×10^6 N/m² and stagnation temperatures T_0 from 317° to 442° K).

31. Jones, R. A.; and Richie, C. B.: Effect of Wall Temperature Variations on Aerodynamic Heating to an Actively Cooled Aircraft. *Journal of Aircraft*, vol. 11, no. 8, August 1974, pp. 488-489.

Prior studies have indicated the potential advantages of an active cooling system approach over alternate cooling concepts and uncooled structure for liquid hydrogen-fueled hypersonic aircraft. One concept shown to be feasible for Mach numbers up to 8 employs unprotected aluminum alloy structure cooled by a liquid convective system to normal working temperatures of 250° F to 300° F. This concept involves the incorporation of tubular coolant passages in the load carrying skin. Each loop of the coolant distribution system contains a network of lines that connect the airframe skin panels, where heat is absorbed by a secondary water-glycol coolant, to a central heat-exchanger where heat is rejected to the liquid hydrogen fuel on its way to the engine. The structural skin represents a very large surface area from which heat must be removed. Thus, the design of the cooled skin panels, which requires consideration of many parameters such as heat load, skin thickness, panel size, tube spacing, tube size, temperature limits, coolant flow, etc., must minimize weight and thermal stresses.

32. Hunt, James L.; Jones, Robert A.; and Smith, K. A.: Use of Hexafluoromethane to Simulate the Inviscid Real-Gas Effects on Blunt Entry Vehicles. *NASA TN D-7701*, October 1974.

Results are presented of an experimental and theoretical investigation of the use of equilibrium hexafluoroethane (C_2F_6) as an undissociated test gas in a wind tunnel to simulate the inviscid aerodynamic characteristics of blunt bodies in high-speed flight where dissociation occurs. The results indicate that the use of C_2F_6 as a test gas in wind tunnels is a practical and relatively simple and inexpensive method of obtaining test conditions with large normal-shock density ratios which is the necessary condition for blunt-body inviscid similitude. Equations for the thermodynamic and transport properties of C_2F_6 are also included.

33. **Ashby, George C., Jr.: Aerodynamic Testing Technique for Twin Fuselage Models at Hypersonic Speeds. NASA TM X-3196, March 1975.**

A testing technique for obtaining the static aerodynamic characteristics of twin-fuselage configurations at hypersonic speeds by using a conventional single-balance installation has been evaluated. Data from a triple-fuselage model and a single-fuselage model were summed and then halved to obtain the characteristics for a twin-fuselage model of the same scale. The three related models were evaluated experimentally at Mach 20.3 in helium and Mach 6 in air for an angle-of-attack range from -6° to 50° . The Reynolds numbers, based on model length were 1.88 million for the Mach 20.3 tests and 2.55 million for the Mach 6 tests.

34. **Jones, Robert A.; Braswell, Dorothy O.; and Richie, Christine B.: Fail-Safe System for Actively Cooled Supersonic and Hypersonic Aircraft. NASA TM X-3125, January 1975.**

A fail-safe-system concept was studied as an alternative to a redundant active cooling system for supersonic and hypersonic aircraft which use the heat sink of liquid-hydrogen fuel for cooling the aircraft structure. This concept consists of an abort maneuver by the aircraft and a passive thermal protection system (TPS) for the aircraft skin. The abort maneuver provides a low-heat-load descent from normal cruise speed to a lower speed at which cooling is unnecessary, and the passive TPS allows the aircraft skin to absorb the abort heat load without exceeding critical skin temperature. On the basis of results obtained, it appears that this fail-safe-system concept warrants further consideration, inasmuch as a fail-safe system could possibly replace a redundant active cooling system with no increase in weight and would offer other potential advantages.

35. **Hunt, J. L.; Talcott, N. A., Jr.; and Cubbage, J. M.: Scramjet Exhaust Simulation Technique for Hypersonic Aircraft Nozzle Design and Aerodynamic Tests. AIAA 15th Aerospace Sciences Meeting, Los Angeles, California. AIAA Paper 77-82, January 1977.**

Current design philosophy for scramjet-powered hypersonic aircraft results in configurations with the entire lower fuselage surface utilized as part of the propulsion system. The lower aft-end of the vehicle acts as a high expansion ratio nozzle. Not only must the external nozzle be designed to extract the maximum possible thrust force from the high energy flow at the combustor exit, but the forces produced by the nozzle must be aligned such that they do not unduly affect aerodynamic balance. The strong coupling between the propulsion system and the aerodynamics of the aircraft makes imperative at least a partial simulation of the inlet, exhaust, and external flows of the hydrogen-burning scramjet in conventional facilities for both nozzle formulation and aerodynamic-force data acquisition. Aerodynamic testing methods offer no contemporary approach for such vehicle design requirements. NASA-Langley has pursued an extensive scramjet/airframe integration R&D program for several years and has recently developed a promising technique for simulation of the scramjet exhaust flow for hypersonic aircraft. Current results of the research program to develop a scramjet flow simulation technique through the use of substitute gas blends are described in this paper.

36. **Talcott, Noel A., Jr.: Thermodynamic Properties of Gaseous Fluorocarbons and Isentropic Equilibrium Expansions of Two Binary Mixtures of Fluorocarbons and Argon. NASA TN D-8405, April 1977.**

Equations and computer code are given for the thermodynamic properties of gaseous fluorocarbons in chemical equilibrium. In addition, isentropic equilibrium expansions of two binary mixtures of fluorocarbons and argon are included. The computer code calculates the equilibrium thermodynamic properties and, in some cases the transport properties for the following fluorocarbons: CCl_3F , CCl_2F_2 , $CBrF_3$, CF_4 , $CHCl_2F$, CHF_3 , CCl_2F-CCl_2F , $CClF_2-CClF_2$, CF_3-CF_3 , and C_4F_8 . Equilibrium thermodynamic properties are tabulated for six of the fluorocarbons (CCl_3 , CCl_2F_2 , $CBrF_3$, CF_3-CF_3 , and C_4F_8) and pressure-enthalpy diagrams are presented for $CBrF_3$.

37. **Talcott, N. A., Jr.; and Hunt, J. L.: Streamtube Analysis of a Hydrogen-Burning Scramjet Exhaust and Simulation Technique. Journal of Aircraft, Vol. 14, No. 9, September 1977.**

This note presents the results of a study to determine if this simulation technique can be extended to more complicated flows approaching the complexity of the actual exhaust flow. An analysis is made to determine the state of the flow and the accuracy of the substitute gas simulation in the presence of a shock discontinuity.

38. **Helms, Vernon T.: Measurement of Flow Angle and Mass Flow Rate in an Unknown Flow Field Using Hot Film and Hot Wire Probes. AIAA Paper 78-794, April 1978.**

A data reduction technique applicable to constant-temperature hot-film and hot-wire probes is presented which is used to determine flow angle and mass flow rate in an unknown flowfield over a wide range of flow conditions at supersonic and hypersonic velocities. The technique virtually eliminates the effects of Reynolds number on a probe's flow angle sensitivity. Methods for extrapolating a limited amount of mass flow rate calibration data to include the range of mass flow rate encountered in an experiment are also given. This technique has been applied to data obtained using a hot-film probe during surveys in the leeside flowfield of a Space Shuttle Orbiter configuration.

39. **Helms, Vernon T. III: Measuring Flow Angle and Mass Flow Rate in an Unknown Flowfield. Journal of Spacecraft and Rockets, vol. 16, no. 1, January-February 1979, pp. 20-26.**

A data reduction technique applicable to constant temperature hot film and hot wire probes is presented which is used to determine flow angle and mass flow rate in an unknown flow field over a wide range of flow conditions at supersonic and hypersonic velocities. The technique virtually eliminates the effect of Reynolds number on a probe's flow angle sensitivity. Methods for extrapolating a limited amount of mass flow rate calibration data to include the range of mass flow rate encountered in an experiment are also given. This technique has been applied to data obtained using a hot film probe during surveys in the leeside flow field of a space shuttle orbiter configuration.

40. **Hopkins, H. B.; Konopka, W. and Leng, J.: Validation of Scramjet Exhaust Simulation Technique at Mach 6. NASA CR-3003, March 1979.**

Current design philosophy for hydrogen-fueled, scramjet-powered hypersonic aircraft results in configurations with strong couplings between the engine plume and vehicle aerodynamics. The experimental verification of the scramjet exhaust simulation is described. The scramjet exhaust was reproduced for the Mach 6 flight condition by the detonation tube simulator. The exhaust flow pressure profiles, and to a large extent the heat transfer rate profiles, were then duplicated by cool gas mixtures of Argon and Freon 1381 or Freon 12. The results of these experiments indicate that a cool gas simulation of the scramjet exhaust is a viable simulation technique except for phenomena which are dependent on the temperature relative to flow temperature.

41. **Miller, Charles G.: Comparison of Thin-Film Resistance with Thin-Skin Transient Calorimeter Heat Transfer Gages in Conventional Wind Tunnels. NASA TM-83197, November 1981.**

Thin film gages deposited at the stagnation region of small (8.1-mm-diameter) hemispheres and gages mounted flush with the surface of a sharp-leading-edge flat plate were tested in the Langley continuous-flow hypersonic tunnel and in the Langley hypersonic CF4 tunnel. Two substrate materials were tested, quartz and a machinable glass-ceramic. Small hemispheres were also tested utilizing the thin-skin transient calorimeter technique usually employed in conventional tunnels. One transient calorimeter model was a thin shell of stainless steel, and the other was a thin-skin insert of stainless steel mounted into a hemisphere fabricated from a machinable-glass-ceramic. Measured heat-transfer rates from the various hemispheres were compared with one another and with predicted rates. The results demonstrate the feasibility and advantages of using film resistance heat transfer gages in conventional hypersonic wind tunnels over a wide range of conditions.

42. **Buck, Gregory M.: Aerodynamic Simulation and Scaling of Drag in Hypersonic Low Density Flows. AIAA Paper 86-0743-OP, March 1986.**

SUBJECT CATEGORY 9 - AIRFRAME ACTIVE COOLING SYSTEMS

1. **Ferri, A.; Fox, H.; and Hoydysh, W.: Turbine-Compressor System for Active Cooling of Hypersonic Aircraft. NASA CR-66930, January 1970.**
2. **McConarty, W. A.; and Anthony, F. M.: Design and Evaluation of Active Cooling Systems for Mach 6 Cruise Vehicle Wings. NASA CR-1916, December 1971.**

Active cooling systems, which included transpiration, film, and convective cooling concepts, are examined. Coolants included hydrogen, helium, air, and water. Heat shields, radiation barriers, and thermal insulation are considered to reduce heat flow to the cooling systems. Wing sweep angles are varied from 0° to 75° and wing leading edge radii of 0.05 inch and 2.0 inches are examined. Structural temperatures are varied to allow comparison of aluminum alloy, titanium alloy, and superalloy structural materials. Cooled wing concepts are compared among themselves, and with the uncooled concept on the basis of structural weight, cooling system weight, and coolant weight.

3. **Helenbrook, R. G.; McConarty, W. A.; and Anthony, F. M.: Evaluation of Active Cooling Systems for a Mach 6 Hypersonic Transport Airframe. NASA CR-1917, December 1971.**

Transpiration and convective cooling concepts are examined for the fuselage and tail surfaces of a Mach 6 hypersonic transport aircraft. Hydrogen, helium, and air are considered as coolants. Heat shields and radiation barriers are examined to reduce heat flow to the cooled structures. The weight and insulation requirements for the cryogenic fuel tanks are examined so that realistic totals can be estimated for the complete fuselage and tail. Structural temperatures are varied to allow comparison of aluminum alloy, titanium alloy, and superalloy construction materials. The results of this study are combined with results obtained on the wing structure, obtained in a previous study, to estimate weights for the complete airframe. The cooled concepts are compared among themselves, and with the uncooled concept on the basis of structural weight, cooling system weight, and coolant weight.

4. **Helenbrook, R. G.; and Anthony, F. M.: Design of a Convective Cooling System for a Mach 6 Hypersonic Transport Airframe. NASA CR-1918, December 1971.**

Results of analytical and design studies are presented for a water-glycol convective cooling system for the airframe structure of a hypersonic transport. System configurations and weights are compared. The influences of system pressure drop and flow control schedules on system weight are defined.

5. **Piva, R.: Leading-Edge Cooling System by Upstream Injection Along Surface. NASA CR-111965, 1971.**

A leading-edge cooling system by upstream injection along the surface was investigated. The purpose of this system is to keep the leading edge below a desired temperature without excessively increasing the radius of the tip and consequently the total pressure losses. An experimental investigation was conducted to find the optimum conditions for the cooling from the point of view of upstream jet penetration and minimum shock losses. A theoretical analysis was performed to study the flow field in the mixing region between the two counterflowing streams and the results obtained compare favorably with the experimental results.

6. **Wagner, R. D.: Optimal Skin Thickness Variations for Active Cooled Panels. Journal of Aircraft, vol. 10, no. 9, September 1973, pp. 572-573.**

7. **Jones, R. A.; Braswell, D. O.; and Richie, C. B.: Fail-Safe System for Actively Cooled Supersonic and Hypersonic Aircraft. NASA TM X-3125, January 1975.**

A preliminary study was made of a fail-safe-system concept as an alternative to a redundant active cooling system for supersonic and hypersonic aircraft which use the heat sink of liquid-hydrogen fuel for cooling the aircraft structure. This concept consists of an abort maneuver by the aircraft and a passive thermal protection system (TPS) for the aircraft skin. The abort maneuver provides a low-heat-load descent from normal cruise speed to a lower speed at which cooling is unnecessary, and the passive TPS allows the aircraft skin to absorb the abort heat load without exceeding critical skin temperature.

8. **Pirrello, C. J.; and Herring, R. L.: Study of a Fail-Safe Abort System for an Actively Cooled Hypersonic Aircraft - Vol. II. NASA CR-144920, January 1976.**

Conceptual designs of a fail-safe abort system for hydrogen-fueled actively cooled high-speed aircraft are examined. The fail-safe concept depends on basically three factors: (1) a reliable method of detecting a failure or malfunction in the active cooling system, (2) the optimization of abort trajectories which minimize the descent heat load to the aircraft, and (3) fail-safe thermostructural concepts to minimize both the weight and the maximum temperature the structure will reach during descent. These factors are examined and promising approaches are evaluated based on weight, reliability, ease of manufacture and cost.

9. **Herring, R. L.; and Stone, J. E.: Thermal Design for Areas of Interference Heating on Actively Cooled Hypersonic Aircraft. NASA CR-2828, December 1976.**

Numerous actively cooled panel design alternatives for application in regions on high-speed aircraft that are subject to interference heating effects were studied. Candidate design concepts were evaluated using mass, producibility, reliability, and inspectability/maintainability as figures of merit. Three design approaches were identified as superior within certain regimes of the matrix of design heating conditions considered. Only minor modifications to basic actively cooled panel design are required to withstand minor interference heating effects. Designs incorporating internally finned coolant tubes to augment heat transfer are recommended for moderate design heating conditions.

10. **Pagel, L. L.; and Herring, R. L.: Evaluation of a Large Capacity Heat Pump Concept for Active Cooling of Hypersonic Aircraft Structure. NASA CR-145301, February 1978.**

Results of engineering analyses assessing the conceptual feasibility of a large capacity heat pump for enhancing active cooling of hypersonic aircraft structure are presented. A unique heat pump arrangement which permits cooling the structure of a Mach 6 transport to aluminum temperatures without the aid of thermal shielding is described. The selected concept is compatible with the use of conventional refrigerants, with Freon R-11 selected as the preferred refrigerant. Condenser temperatures were limited to levels compatible with the use of conventional refrigerants by incorporating a unique multipass condenser design, which extracts mechanical energy from the hydrogen fuel, prior to each subsequent pass through the condenser.

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